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FINAL PROGRESS REPORT

BIORESEARCH MODULE DESIGN DEFINITION AND SPACE SHUTTLE VEHICLE INTEGRATION STUDY

REPORT NO. T146-4
17 DECEMBER 1971

VOLUME II - APPENDICES

SPONSORED BY
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
AMES RESEARCH CENTER
MOFFETT FIELD, CALIFORNIA 94035

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this document may be better
studied on microfiche

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VOLUME II - APPENDICES

By A. L. Lang, Jr.

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Prepared under Contract No. NAS2-6524 by
VOUGHT MISSILES AND SPACE COMPANY

LTV AEROSPACE CORPORATION

Dallas, Texas 75222

for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

FOREWORD

This Final Progress Report for the "Bioresearch Module Design Definition and Space Shuttle Vehicle Integration Study", NASA Contract NAS2-6524, is provided in accordance with Article IV of the contract schedule. The six-month period of performance of this contractual work was 4 June 1971 through 4 December 1971. During this period two Design Reviews were conducted, the first at the Grand Prairie, Texas facility of Vought Missiles and Space Company, LTV Aerospace Corporation; the second at the National Aeronautics and Space Administration, Ames Research Center, Moffett Field, California. These reviews are reported in separate volumes:

"Bioresearch Module First Design Review," 26, 27 August 1971,
LTV/VMSC Report No. T146-2

"Bioresearch Module Second Design Review," 5 November 1971,
LTV/VMSC Report No. T146-3.

The material presented at the Design Reviews plus the results of all contractual studies are included in this Final Progress Report. The report is submitted in two volumes:

Volume I - Basic Report

Volume II - Appendices

Cost analyses presented in Sections 4.6 and 4.7 of Volume I are submitted as separate attachments.

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APPENDIX A

SUMMARY DESCRIPTION OF POWER SYSTEM

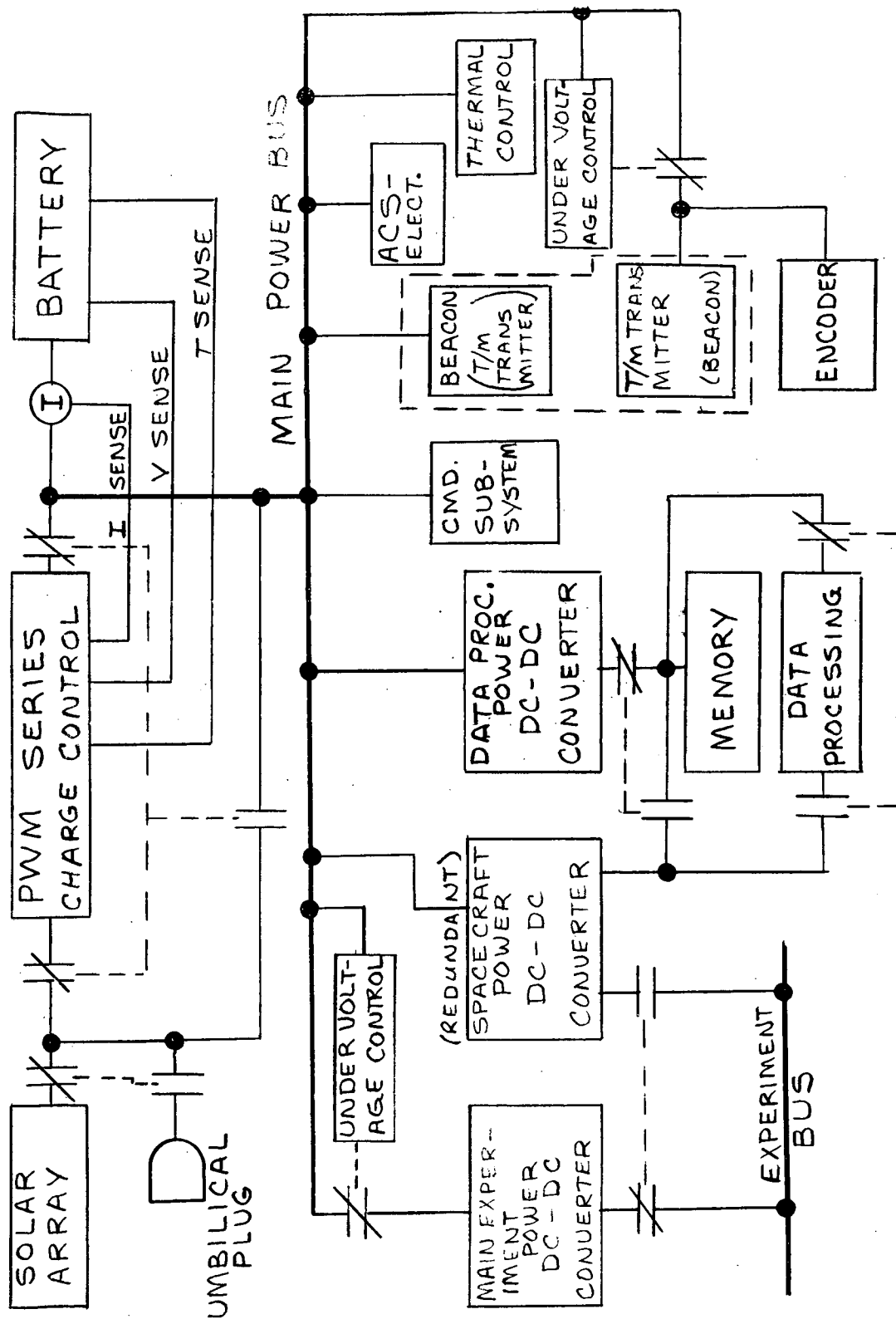


FIGURE A-1. - POWER SYSTEM DIAGRAM

SUMMARY DESCRIPTION OF POWER SYSTEM

The following is a description and performance summary of the electrical power subsystem of the Bioresearch Module spacecraft.

1.0 POWER SYSTEM

The subsystem is powered by a solar array-rechargeable battery with series Pulse Width Modulated (PWM) charge control. Figure A-1 is a block diagram of the power system for Missions I, I(S), and II. The diagram for Mission III would differ in that the beacon and T/M transmitter are replaced by a GRARR VHF transponder. The system employs separate power converters for the experiments and the spacecraft systems. A redundant power converter (spacecraft power) is included to supply either or both if the respective converters fail. The use of the two converters (memory and main experiment) allows a greater overall conversion efficiency than would the use of one converter for both functions. The efficiency increase stems from the higher tolerances on the experiment-required voltages. Table A-1 summarizes experiment and data processing conversion efficiencies and conversion techniques. The efficiencies shown are based on average nominal loading.

Two T/M transmitters are used. One serves as a beacon and the other, the transmitter. Each is the redundant of the other.

TABLE A-1. - POWER CONVERTER EFFICIENCY

<u>VOLTAGE</u>	<u>EFFICIENCY</u>	<u>REGULATED BY</u>
<u>EXPERIMENT POWER CONVERTER</u>		
27.5 \pm 2.5	95%	Using a limited range series regulator - resulting in a \pm 2V variation due to load and battery voltage variation.
+15 \pm 2.5	83%	Feedback from +5V output-resulting in less than \pm 1V variation due to load and battery voltage variation
+5 \pm .02	76%	Preregulator feedback.
-15 \pm .5	70%	Post regulator (Low voltage drop series regulator)
<u>DATA PROCESSING PWR. CONVERTER</u>		
+15 \pm .75	70%	Post Regulator
+5 \pm .25	75%	Preregulator Feedback
-5 \pm .10	70%	Post Regulator

2.0 POWER REQUIREMENTS

2.1 SPACECRAFT POWER REQUIREMENTS

The requirements for each mission are shown in Tables A-2 through A-5 with efficiencies included. The values under the column labeled "continuous" are the standby power requirements of each component that either must be energized continuously such as the memory or have a 100% duty cycle such as the spacecraft clock. The column labeled " Δ PWR" is the increase in power by each component when in the "operate" mode. The column labeled "duty cycle" contains the percent of mission time each component is in the "operate" mode; e.g., the 15% duty cycle of the encoder indicates that for each 96-minute orbit, the encoder will operate for a total of 14.4 minutes. The column labeled "Avg. PWR" contains the average power demand in watts of each component. The values are derived as follows:

$$\text{Avg. PWR} = [\text{Continuous Power} + (\frac{\text{Duty Cycle}}{100\%})(\Delta \text{PWR})] \text{ watts}$$

The total average spacecraft power demand for each mission is as follows:

I	16.7	
I(S)	18.1	
		watts
II	14.3	
III	31.0	

The peak power of each mission is as follows:

I	68.53	
I(S)	140.73	
		watts
II	75.03	
III	77.05	

and is found by summing the "continuous" and " Δ PWR" columns.

It should be noted that an S-band transmitter requirement was added to Mission I(S) to show that the system could handle a high powered, low duty cycle component. With this addition, the Δ PWR of data processor increased from 3.50 to 4.00 watts. The S-band power demand is carried throughout the analysis in Mission I(S) only.

TABLE A-2. - MISSION I POWER REQUIREMENTS
(EFFICIENCIES INCLUDED)

	CONTINUOUS	Δ PWR.	DUTY CYCLE	AVG. PWR.
Beacon	0.70	-	-	0.70
T/M Transmitter	-	0.70	12%	0.09
Encoder	-	8.60	15%	1.30
Housekeeping Sensors	-	0.15	15%	0.02
Signal Conditioners	-	1.30	15%	0.20
Memory	0.65	1.25	15%	0.83
Data Processing	0.80	3.50	15%	1.33
S/C Clock	0.25	-	-	0.25
Command Receiver	2.14	2.44	12%	2.43
Command Decoder	0.25	6.00	12%	1.00
Thermal System	0.70	-	-	0.70
Attitude Control	-	-	-	-
(a) Reaction Control	-	13.10	.05%	0.01
(b) Rate Gyro	-	15.00	2%	0.30
(c) Integrating Rate Gyro	-	6.00	40%	2.40
(d) Control Electronics	5.00	-	-	5.00
(e) Sun Sensors	-	-	-	-
MISSION I AVERAGE				
SPACECRAFT POWER DEMAND				
16.7 WATTS.				
PEAK = 68.3 WATTS				

TABLE A-3. - MISSION I(S) POWER REQUIREMENTS

(EFFICIENCIES INCLUDED)

	<u>CONTINUOUS</u>	<u>ΔPWR.</u>	<u>DUTY CYCLE</u>	<u>AVG. PWR.</u>
Beacon	0.70	-	-	0.70
T/M Transmitter - VHF	-	0.70	12%	0.09
T/M Transmitter - S-Band	-	72.40	2%	1.45
Encoder	-	8.60	15%	1.30
Housekeeping Sensors	-	0.15	15%	0.02
Signal Conditioners	-	1.30	15%	0.20
Memory	0.65	1.25	15%	0.83
Data Processing	0.80	4.00	15%	1.40
S/C Clock	0.25	-	-	0.25
Command Receiver	2.14	2.44	12%	2.43
Command Decoder	0.25	6.00	12%	1.00
Thermal System	0.70	-	-	0.70
Attitude Control	-	-	-	-
(a) Reaction Control	-	13.10	.05%	0.01
(b) Rate Gyro	-	15.00	2%	0.30
(c) Integrating Rate Gyro	-	6.00	40%	2.40
(d) Control Electronics	5.00	-	-	5.00
(e) Sun Sensors	-	-	-	-

MISSION I(S) AVERAGE SPACECRAFT

POWER DEMAND 18.1 WATTS

PEAK = 140.73 WATTS

TABLE A-4. * MISSION II POWER REQUIREMENTS
(EFFICIENCIES INCLUDED)

	<u>CONTINUOUS</u>	<u>Δ PWR.</u>	<u>DUTY CYCLE</u>	<u>AVG. PWR.</u>
Beacon	0.70	-	-	0.70
T/M Transmitter	-	0.70	12%	0.09
Encoder	-	8.60	15%	1.30
Housekeeping Sensors	-	0.15	15%	0.02
Signal Conditioners	-	1.30	15%	0.20
Memory	0.65	1.25	15%	0.83
Data Processing	0.80	3.50	15%	1.33
S/C Clock	0.25	-	-	0.25
Command Receiver	2.14	2.44	12%	2.43
Command Decoder	0.25	6.00	12%	1.00
Thermal System	0.70	-	-	0.70
Attitude Control	-	-	-	-
(a) Reaction Control	-	13.10	.05%	0.01
(b) Rate Gyro	-	15.00	2%	0.30
(c) Horizon Sensors	-	0.50	5%	0.03
(d) Sun Sensors	-	-	-	-
(e) Control Electronics	5.00	-	-	5.00
Booms	-	12.00	1%	.12
MISSION II AVERAGE SPACECRAFT				
POWER DEMAND 14.3 WATTS				
PEAK - 75.03 WATTS				

TABLE A-5. - MISSION III POWER REQUIREMENTS

(EFFICIENCIES INCLUDED)

	<u>CONTINUOUS</u>	<u>Δ PWR.</u>	<u>DUTY CYCLE</u>	<u>AVG. PWR.</u>
Grarr Xponder	20.00	-	-	20.00
Encoder	-	8.60	15%	1.30
Housekeeping Sensors	-	0.15	15%	0.02
Signal Conditioners	-	1.30	15%	0.20
Memory	0.65	1.25	15%	0.83
Data Processing	0.80	3.50	15%	1.33
S/C Clock	0.25	-	-	0.25
Command Decoder	0.25	6.00	12%	1.00
Thermal System	0.70	-	-	0.70
Attitude Control				
(a) Reaction Control	-	13.10	.05%	0.01
(b) Rate Gyro	-	15.00	2%	0.30
(c) Horizon Sensors	-	0.50	5%	0.03
(d) Sun Sensors	-	-	-	-
(e) Control Electronics	5.00	-	0	5.00
MISSION III AVERAGE SPACECRAFT				
POWER DEMAND 31.0 WATTS				
PEAK = 77.05 WATTS				

2.2 EXPERIMENT POWER REQUIREMENTS

Table A-6 shows the baseline and revised experiment power requirements with efficiencies included.

2.3 POWER REQUIREMENTS OF SYSTEM

2.3.1 Experiment and Spacecraft Power. - Table A-7 shows the total power required to operate the spacecraft and experiment for both the baseline and revised power profiles. These values are formed by summing the mission by mission total average power given in Tables A-2 through A-5 with the corresponding experiment average in Table A-6; e.g., for Mission I:

From Table A-2	16.7 watts
From Table A-6	<u>87.1 watts</u>
Baseline Avg. Pwr. (Incl. efficiency)	103.8 watts

2.3.2 Solar Array Power Demand. The Bioresearch Module uses a solar array/rechargeable battery configuration in which the battery operates the spacecraft when in the shadow of the earth; therefore, the array must both power the module and supply enough energy to the battery for "night" operation. Table A-8 shows the solar array demand for each mission. The values for Missions I, I(S), and II are based on a 62% illuminated orbit and for Mission III, a 96% illuminated orbit. Battery charge and discharge efficiencies are included. The technique in deriving these values is as follows:

$$\text{Power Demanded} = \text{Avg. Pwr.} \left[1 + \left(\frac{\text{Dark}}{\text{Light}} \right) \left(\frac{1}{\text{Charge Eff.}} \right) \left(\frac{1}{\text{Dischg. Eff.}} \right) \right]$$

For Missions I, I(S) and II,

$$\text{Power Demanded} = 1.807 \text{ Avg. Pwr. using 62\% illuminated orbit.}$$

For Mission III,

$$\text{Power Demanded} = 1.055 \text{ Avg. Pwr. using a 95\% illuminated orbit.}$$

Where the battery charge efficiency is 95% and discharge efficiency is 80%.

TABLE A-6. - BASELINE AND REVISED EXPERIMENT POWER REQUIREMENTS

Baseline Experiment Power Requirements, All Missions

	Required	With Efficiency
27.5 \pm 2.5 VDC	64 watts continuous 89 watts peak for 6 min/hr }	70 watts Avg.
+15 \pm 2.5 VDC	3 watts continuous	3.6 watts
-15 \pm 0.5 VDC	3 watts continuous	4.3 watts
+5 \pm 0.02 VDC	7 watts continuous	9.2 watts
TOTAL	87.1 watts average (including efficiency)	

Revised Experiment Power Requirements, Missions I, II, III

	Required	With Efficiency
27.5 \pm 2.5 VDC	31 watts continuous 111 watts peak for 3 min/hr }	37 watts Avg.
+15 \pm 2.5 VDC	2 watts continuous	2.4 watts
-15 \pm 0.5 VDC	2 watts continuous	2.8 watts
+5 \pm 0.02 VDC	5 watts continuous	6.6 watts
TOTAL	48.8 watts average (including efficiency)	

Revised Experiment Power Requirements, Mission I(S)

	Required	With Efficiency
27.5 \pm 2.5 VDC	20 watts continuous 45 watts peak for 2 min/orbit 150 watts peak for 15 sec/orbit }	22 Watts Avg.
+15 \pm 2.5 VDC	2 watts continuous	2.4 watts
-15 \pm 0.5 VDC	2 watts continuous	2.8 watts
+5 \pm 0.02 VDC	5 watts continuous	6.6 watts
TOTAL	33.8 watts average (including efficiency)	

TABLE A-7. - EXPERIMENT AND SPACECRAFT PWR.

(EFFICIENCIES INCLUDED)				
	Mission I	Mission I(S)	Mission II	Mission III
Baseline	103.8 watts	105.3 watts	101.5 watts	118.1 watts
Revised (Reference A-6, Item 1)	65.5 watts		63.2 watts	79.8 watts
Revised (Reference A-6, Item 4)		52.0 watt		

TABLE A-8. - SOLAR ARRAY POWER

DEMAND*

	Mission I	Mission I(S)	Mission II	Mission III
Baseline	187 watts	190 watts	183 watts	125 watts
Revised	118 watts		114 watts	85 watts
Revised		94 watts		

*62% Illuminated Orbit, Battery Efficiencies Included.

3.0 SOLAR ARRAY SIZING

The selection of solar cells was made in the Bioexplorer study, Reference A-2, and the cells remain 2 x 2 cm, 12 mil N/P silicon cells.

3.1 SOLAR ARRAY RADIATION ENVIRONMENT AND DEGRADATION

Fused silica cell covers of 25 mil thickness were selected to achieve maximum mission versatility. Table A-9 -shows the expected environment with these covers. The 12 mil cover is shown for comparison.

With the radiation environment defined, the radiation degradation can be calculated by employing the following equation:

$$Q = \left[\frac{7}{9} \left(\frac{\phi}{\phi_c} \right)^{1/2} + 1 \right]^{-1/2}$$

Where

Q = power remaining
 ϕ = integrated particle flux
 ϕ_c = critical flux = 2×10^{11} protons/cm²
 = 6×10^{14} electrons/cm²

The following results were obtained:

For Missions I, I(S), and II,

25 mil	6% degradation
12 mil	8%

For Mission III,

25 mil	20%
12 mil	31%

Unfortunately, radiation is not the only mechanism for degradation of the cells. Other mechanisms include cell mismatch, string mismatch, physical damage, cell cover impurities, and spacecraft alignment with respect to the sun. It should be noted that, except for some cover discoloring and radiation, these degradation factors occur on the ground. For the purpose of analysis, all degradation except radiation is included in the following sections. Table A-10 shows the expected degradation from these factors and from radiation.

3.2 SOLAR ARRAY CONCEPTS AND REQUIREMENTS

The Bioresearch Module will use a body-mounted, fixed array consisting of 120° segments wrapped around the spacecraft for Missions II and III, and a combination of segments and deployable panels

TABLE A-9. - SOLAR CELL RADIATION ENVIRONMENT

	25 Mil Cover		12 Mil Cover	
Proton Cutoff	10 MeV		6.6 MeV	
Electron Cutoff	500 KeV		320 KeV	
Particles During Mission Life				
	Protons/CM ²		Electrons/CM ²	
	25 Mil	12 Mil	25 Mil	12 Mil
Mission I, II	3x10 ⁹	4x10 ⁹	5x10 ¹¹	2x10 ¹²
Mission III	6x10 ¹⁰	2x10 ¹¹	5x10 ¹²	10 ¹³

TABLE A-10. - SOLAR CELL DEGRADATION

<u>MECHANISM</u>	<u>MISSIONS I & II</u>	<u>MISSION III</u>
1. Cell Mismatch	1%	1%
2. String Mismatch	2%	2%
*3. Cell Covers (25 Mil)	6%	6%
4. Spacecraft Alignment	1%	1%
5. Radiation (6 Mo.)	<u>6%</u>	<u>20%</u>
TOTAL	16%	30%
*Includes 2% Physical Damage		

for Missions I and I(S).

3.2.1 Mission Array Concepts. - Solar array considerations are as follows:

Missions I and I(S). - Missions I and I(S) are non-spinning spacecraft orientated such that the projection of the solar cells is normal to the sun. Body-mounted solar arrays are constrained by both the size and geometry of the vehicle. Since the experiment envelope is smaller for Mission I than for Mission I(S), an analysis of Mission I will be used for both I and I(S). By using the Mission I configuration, a guideline for interchangeability is established.

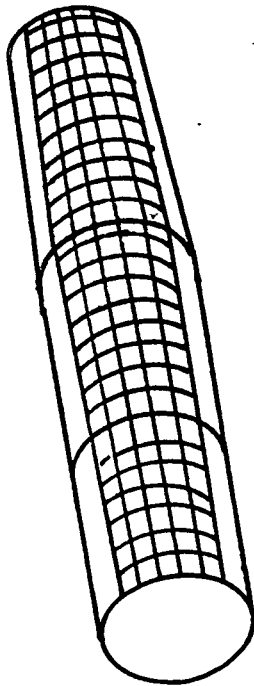
Figure A-2 shows the general geometry of Mission I. Figure A-2a shows the number of solar cells that can be placed on the configuration versus the sector size in degrees. Each sector includes the length of the spacecraft. For a 160° sector, approximately 5000 cells can be employed. The number of cells includes an 85% packing efficiency.

Figure A-2b illustrates the output in watts of a given sector versus time in sunlight. The decay is caused by increased cell temperature. The array output includes all degradation except the 6% radiation shown in Table A-10.

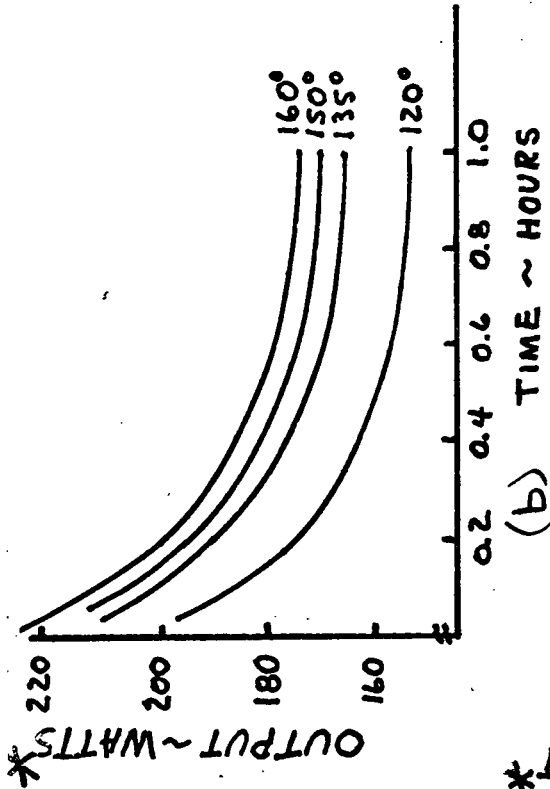
Figure A-2c shows the beginning-of-life (BOL) average output of the array versus sector size. The 180° sector provides an end-of-life (EOL) power of 178 watts (EOL power = BOL power x 0.94 = 189 x 0.94 = 178 watts). Since the baseline Mission I average power demand is 187 watts, the fixed solar array has inadequate power. An alternate array concept for Mission I and I(S) is shown on Figure A-3. The array will consist of 120° segments. Two segments are used to form a fold-out panel unit labeled D. Segments A, B, and C may be utilized. The graph shows 277 watts average BOL output for all segments, and the segment power output and number of cells is noted. All or any combination of the segments may be used as mission power requirements dictate. This will be discussed in section 3.2.2. Figure A-4 illustrates proposed panel deployment concepts.

Missions II and III. - For maximum versatility, the module geometry is divided into three longitudinal segments - the experiment bay, the equipment bay, and the lower unit. Each segment is divided into three 120° sectors giving a total of nine separate solar cell panels which can be added or removed as needed to supply the power required for a given mission. Figure A-5 shows this concept for Mission II. This configuration yields 218 watts average BOL with all degradation except 6% radiation included. Figure A-6 illustrates the sector-segment concept for Mission III. Note that the average BOL power is 221 watts, slightly higher than for Mission II with identical arrays. The difference is due to a lower temperature profile for Mission III due to less heat input from earth albedo.

* ALL DEGRADATION EXCEPT 6% RADIATION INCLUDED



TOTAL SPACECRAFT BASELINE DEMAND : 187 WATTS



* INITIAL AVE. OUTPUT

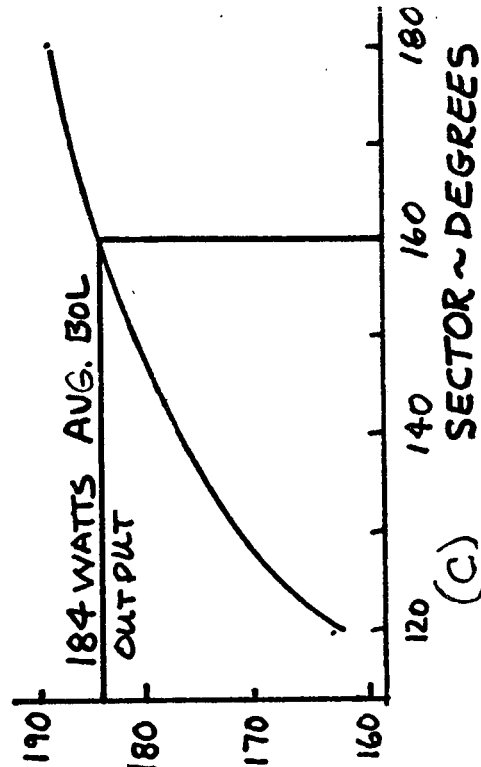
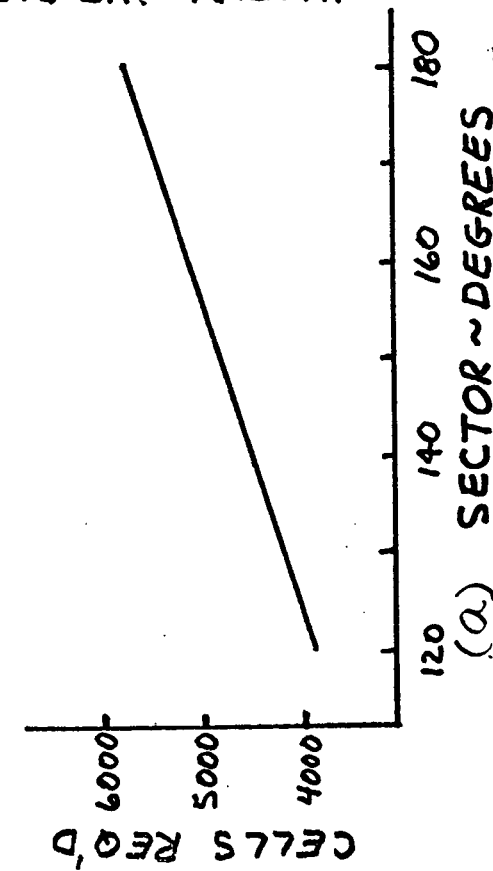
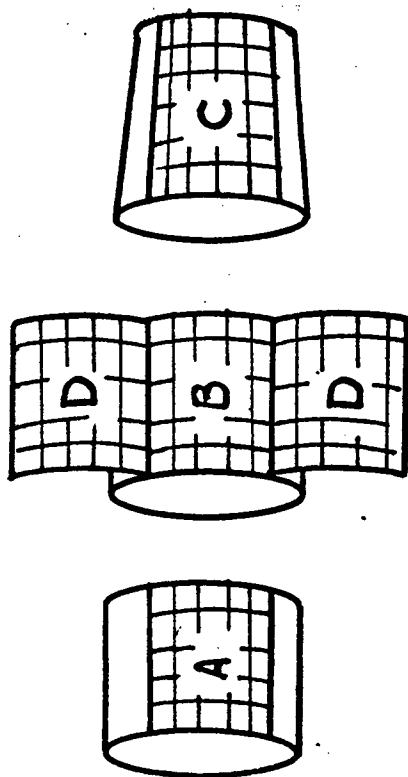
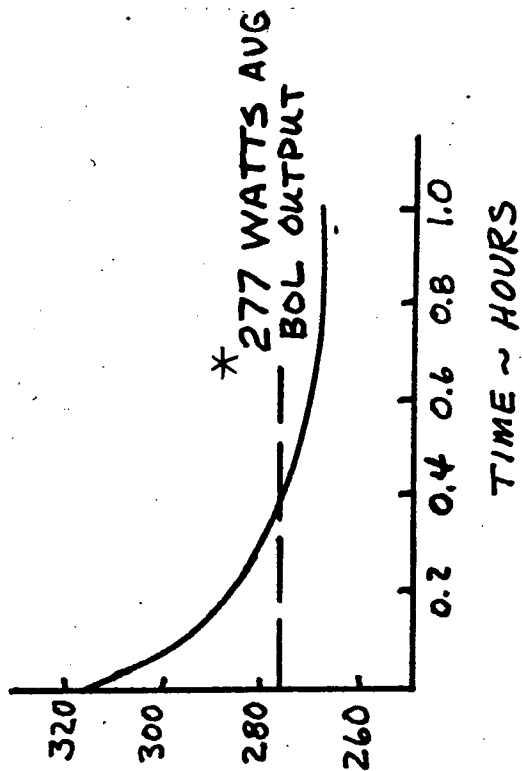


FIGURE A-2.- FIXED ARRAY CONCEPT - MISSION I

* INITIAL POWER ~ WATTS



EACH PANEL IS A
120° SECTOR

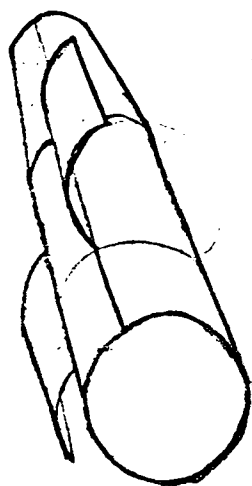


SEGMENT	No. CELLS	OUTPUT (WATTS)
A	1190	48
B	1266	54
C	1380	60
D	2534	115
	<u>6370</u>	<u>277</u>

* ALL DEGRADATION
EXCEPT 6% RADIATION
INCLUDED.

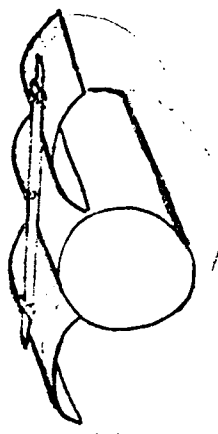
A-17 FIGURE A-3.- ALTERNATE ARRAY CONCEPT - MISSION I

<u>REQUIREMENTS:</u>	
120° Rotation	
Activated by Yo-Yo Release	
38 x 25 Panel	
8.71 Lb. Total	
Solar Cells	3.96 Lb.
Structure	4.75 Lb.

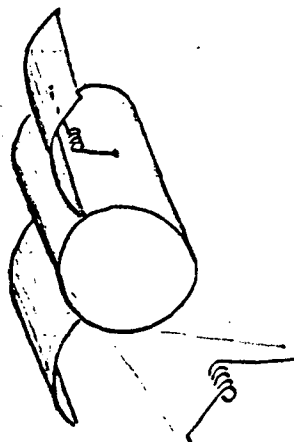


CONCEPTS:

Leaf Spring



Coil Spring



Toggle Mechanism

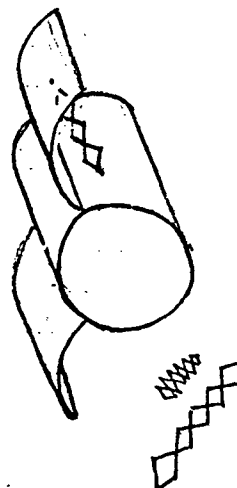
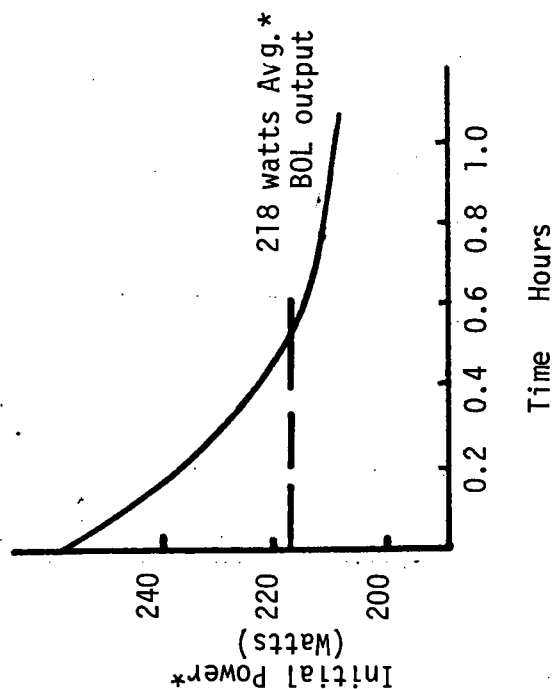
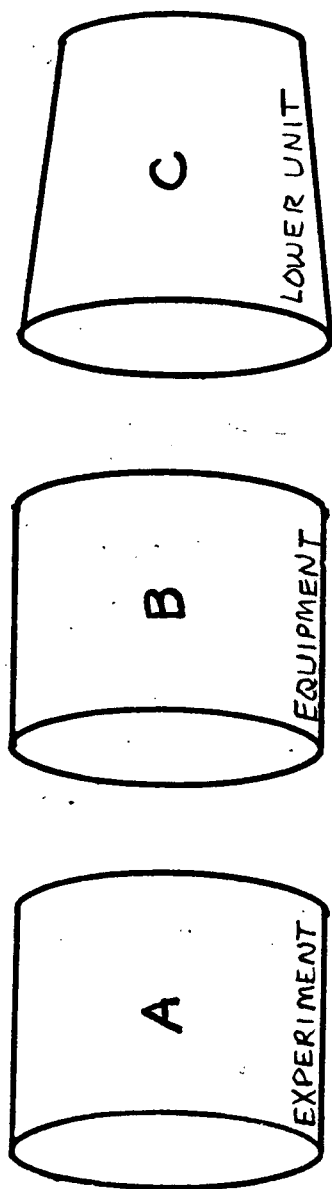


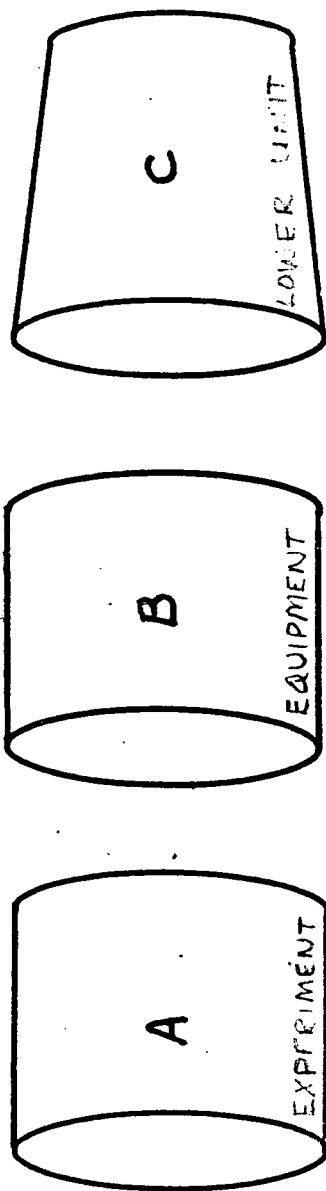
FIGURE A-4. - SOLAR PANEL DEPLOYMENT CONCEPTS



Segment	No. Cells	Output* (Watts)
A	3570	67.5
B	3800	72.0
C	<u>4140</u>	<u>78.5</u>
	11,510	218.0
Total baseline demand:		183 Watts

*All degradation except 6% radiation included

FIGURE A-5. - ARRAY INITIAL OUTPUT - MISSION II



Solar Cell Array
Temperature is
Constant 51°F
for Mission III

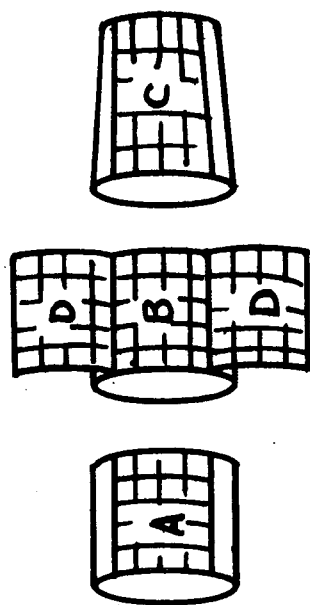
Segment	No. Cells	Output (Watts)
A	3570	68.5
B	3800	73.0
C	<u>4140</u>	<u>79.5</u>
Total Baseline Demand:	11,510	221.0*
125 Watts		

*All degradation except 20% radiation included.

FIGURE A-6. - ARRAY INITIAL OUTPUT - MISSION III

3.2.2. Mission Array Requirements. Having established solar array possibilities and average power available, the next step is to select array size to meet mission power requirements. Table A-8 lists the power demanded by each mission for baseline and revised power profiles, and Figures A-3, A-5, and A-6 define the total power available from the candidate array. It is noted that not all the array area available is necessary for Missions I and I(S). Figure A-7 gives the area or sectors required by the Mission I and I(S) systems. For example, Mission I baseline average power required is 187 watts. By employing sectors A and B and fold-out panels D, the initial (BOL) power is 217 watts, and the final (EOL) power is 204 watts (includes 6% radiation degradation). This gives a 9% end-of-life (EOL) margin. Note that the revised power profile permits elimination of fold-out panels.

In a similar manner, Figure A-8 shows the area required and EOL margin for missions II and III.

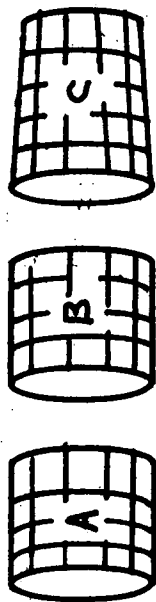


Each segment consists of 120° sectors.

MISSION	REQ'D AREA	NO. SQ. FT.	NO CELLS	BOL * PWR.	EOL PWR.	REQ'D PWR.	EOL MARGIN (%)
I - Baseline	A, B, D	25.3	4990	217	204	187	9
I - Revised	A, B, C.	19.5	3836	162	152	118	29
I(S) - Baseline	A, B, D	25.3	4990	217	204	190	7
I(S) - Revised	B, C	13.4	2646	114	107	94	14

*Includes all degradation except 6% radiation.

FIGURE A-7. - ARRAY REQUIREMENTS - MISSIONS I & I(S)



Each segment is divided into three 120° sectors.

MISSION	REQ'D AREA	NO. SQ. FT.	NO CELLS	BOL * PWR	EOL PWR	REQ'D PWR	EOL MARGIN (%)
II - Baseline	A, B, C	58.4	11,510	218	205	183	12
II - Revised	A, B,	37.4	7,370	139	131	114	15
III - Baseline	1/3 A, B, C	46.3	9,130	175	140	125	12
III - Revised	A, 2/3 B	31.0	6,103	117	94	85	11

*Includes all degradation except radiation.

FIGURE A-8. - ARRAY REQUIREMENTS MISSIONS II & III

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4.0 BATTERY SIZING

As discussed in Reference A-2, the nickel-cadmium battery is selected for the Bioresearch Module. Table A-11 illustrates the sizing of the 12 AH battery and compares it with a 6 AH battery. The latter has inadequate life for baseline requirements.

TABLE A-11. - BATTERY SIZING

MAXIMUM BATTERY LOAD DURING DARK

Baseline: 120 Watts @ 4.4 Amps.

Revised Profile: 82 Watts @ 3.0 Amps

*Assuming 95% discharge efficiency and an 80% charge efficiency at 70°F

	<u>Battery</u>	
	12 AH	6 AH
Depth of Discharge - Baseline	25%	50%
Depth of Discharge - Revised	18%	36%
*No. of Reliable Cycles Available - Baseline	8,000	1,500
*No. of Reliable Cycles Available - Revised	12,000	3,500

12 AH - 23 CELL NICKEL CADMIUM

BATTERY IS ADEQUATE FOR ALL MISSIONS

*References A-1 and A-2.

5.0 COMPONENT SELECTION

There are three categories of electrical components on the Bioresearch Module:

- (1) An Off-the-shelf purchase item, or a GFE item.
- (2) A modified off-the-shelf purchase item, that is, one that required modifications to meet Bioresearch Module requirements without having to be requalified.
- (3) A new fabricated item. An item whose function is unique to the Bioresearch Module such as the Attitude Control Electronics package.

The requirements for selection of off-the-shelf components were as follows:

- (1) Meet the functional and physical requirements of Bioresearch Module.
- (2) Where possible, qualified to the Bioresearch Module environments.
- (3) Where possible, previously flown on similar craft with favorable histories.
- (4) Available at reasonable cost.

The following data was generated for each component:

- (1) Input power
 - (a) Voltage range and tolerance
 - (b) Current range and tolerance
- (2) Unit lifetime and specific limitations (Will the unit operate for six months in space environment for a specified duty cycle?)
- (3) Availability, cost, and delivery time in 1973
- (4) Qualification status
- (5) Specific data unique to the particular component

(6) General information

- (a) History of past and current use
- (b) General description
- (c) Physical description

The components are listed in the following section.

6.0 BIORESEARCH MODULE ELECTRICAL COMPONENTS

The components are listed in item number order. Flight hardware items are numbered on drawings, Figures 1 through 4, Volume I.

ITEM 2 ATTITUDE CONTROL ELECTRONICS

Mission I & I(S)	Mission II	Mission III
<ul style="list-style-type: none">• Monitors 2 Sun sensors, detects errors, energizes thruster through valve amp.• Monitors rate sensors• Cuts on/off integ. gyros• No Gnd. Commands Ref. A-2, Fig. 70, P. 140	<ul style="list-style-type: none">• Control Booms on Cmd.• Processes signals from horizon sensors -• Activates horizon sensors• Commanded from ground• Monitors the sun sensor• Amplifies signal to the six thrusters (Valve amplifier)• Monitors rate sensors Ref. A-2, Fig. 71, p. 146	

The valve amplifier is an integral part of the ACE.

Duty Cycle: Continuous Operation (100%)

Input: 5.00 watts at 28 VDC (incl. eff.)

Qualification: VMSC's past experience on SDP, ALVRJ, and Scout
A new fabricated item.

ITEM 5 THRUSTOR VALVE - Mission I and I(S)

2 Units on Bioresearch Mission I
Sterer P/N 29210, Sterer Engr. & Manuf. Co., Off-the-shelf.

Input: 23.4 ma at 28 VDC
6.55 Watts

Output: Each unit consists of 3 nozzles - A, B, & C
Nozzles A & C output thrust is $-.028 \pm .001$ lbs. each
Nozzles B output thrust is $0.05 \pm .0025$ lbs.

Duty Cycle: Each thruster will be operated 1 time.
Each minute for 30 millisec. duration.

Qualification: Similar to valves used on OA0.

Availability: Will be available for fall, 1973.

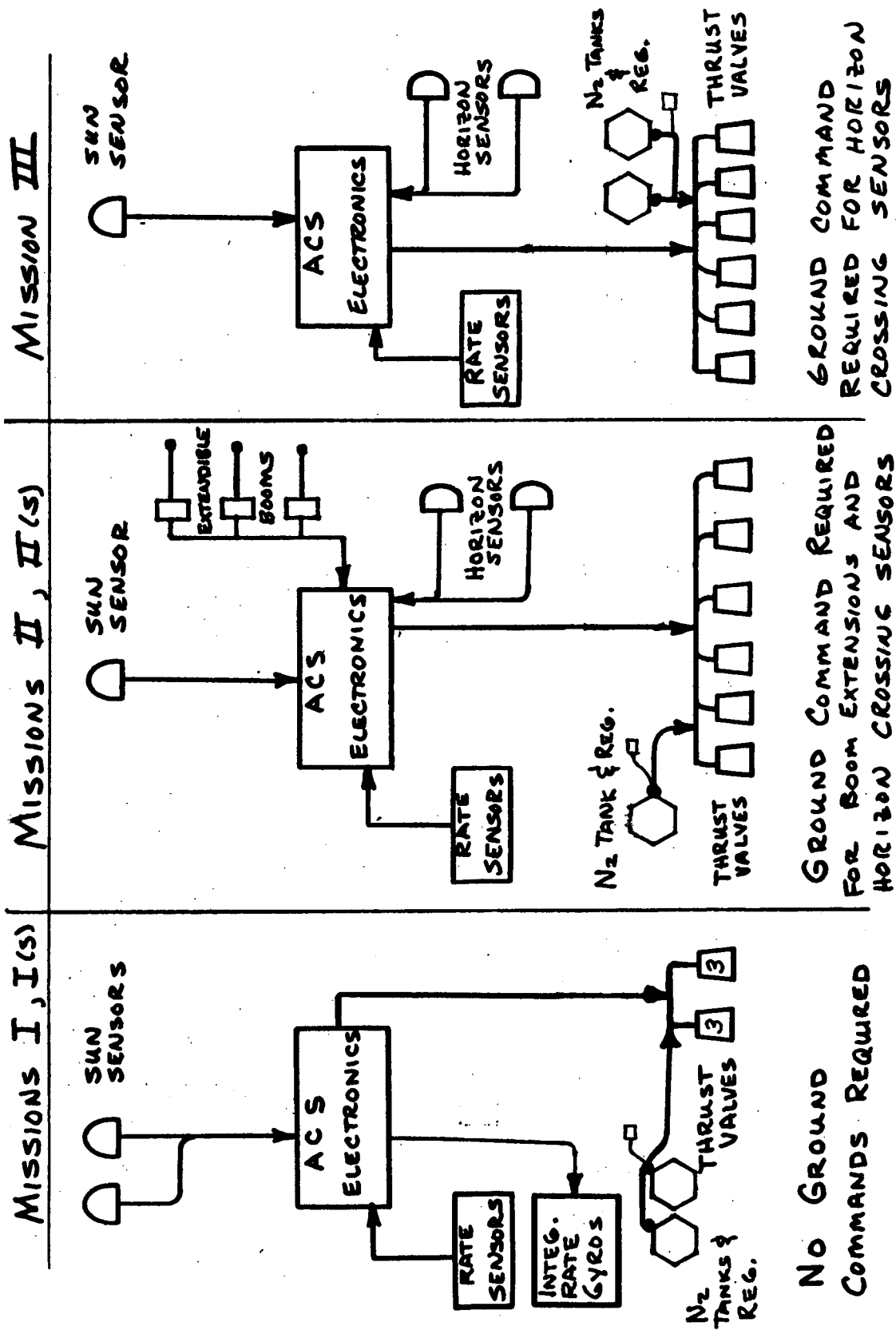


FIGURE A-9.—ATTITUDE CONTROL SYSTEM—COMPONENT DIAGRAM

ITEM 5 THRUSTOR VALVE - Missions II & III

6 units on each Mission II and III

Sterer P/N 24060-1, Modified Off-the-shelf by reducing thrust from 2.5 lbf to 1.0 lbf.

Input: 34.5 watts at 28 VDC each.

Duty Cycle: Random 20 millisec pulses as required by vehicle. (<<1%)

Qualification: Similar to valves used on OAO.

Availability: Will be available for fall, 1973.

ITEM 9 SUN SENSOR

2 sensor on Missions I & I(S), 1 sensor on Missions II & III

Bendix P/N 1771858, The Bendix Corp., Off-the-shelf.

Input: No auxiliary power is required for operation.

Output: 0-5 milliamps.

Duty Cycle: Continuous operation during illuminated periods of orbit.

Qualification: 1771858 and similar units are used on ATM

Availability: Will be available for fall, 1973.

ITEM 10 RATE GYRO - All Missions

1 unit on each Bioresearch Mission

Northrop P/N 79157-350, 3 axis DC/DC standard rate

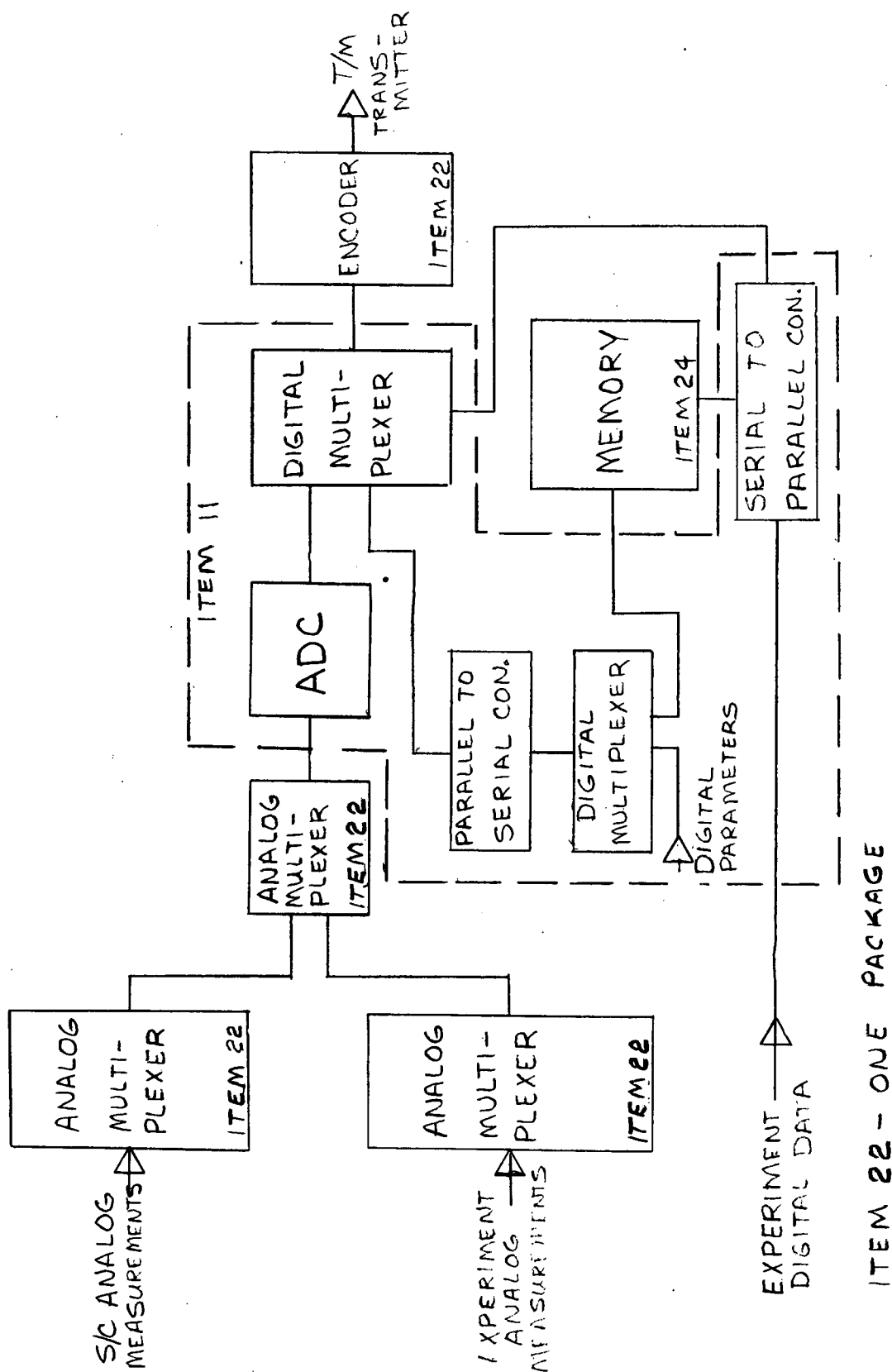
Sensor Assy. Off-the-shelf

Input: 15 watts max. at 28 \pm 3 VDC.

Duty Cycle: The rate gyro package is activated on a low duty-cycle basis (Approx. 2%) to monitor body angular rates.

Qualification: Meets Scout Launch Environment and Spacecraft environment.

Availability: Will be available for fall, 1973.



ITEM 11 -- BASIC DATA PROCESSOR - NEW FAB.

ITEM 16 THERMAL CONTROL ELECTRONICS - All Missions

- 1) (Item 14) Thermistors about the cold plate - monitor cold plate temp. and allow for select cold plate temperatures.(Off-shelf)
- 2) "Set-point" Controller - Biases thermistors to any one of 16 discrete cold plate temperatures. The controller receives signal from ground for selection of temperature. (New fabrication)
- 3) Louver Control Electronics - Monitors thermistor output, when temperature changes the louver actuator is operated to either open or close louvers. (New fabrication.) 2) and 3) are one unit.
- 4) Louver Actuator - Clifton MSL-8-A-1 (Off-the-shelf)

Duty Cycle: 1) & 3) 100% 4) Approx. 3 times per orbit.
2) 15% Louvers are normally closed.

Input: 700 Milliwatts at 15 VDC for system.

ITEM 18 - COMMAND RECEIVER/DEMODULATOR - Missions I & II

2 Units on each Mission I or II Module
SCI Electronics, SCI Proposal P70-1076, New Off-the-shelf

Input: 1.07 watts standby 28 VDC
2.29 watts operate 28 VDC

Duty Cycle: Two Interrogations per orbit.
(12% of the time)

Qualification: Qualified by comparison to existing similar units built by SCI.

Availability: Will be available for fall, 1973.

Very similar to command receiver used on AN/ASW-25.

ITEM 19 COMMAND DECODER - Missions I & II & III

One unit on each Bioresearch Module.
AVCO AED 407 with PCM front end. Off-the-shelf.

Input: 250 milliwatts at 28 VDC standby
6 watts at 28 VDC operate

Duty Cycle: Two Interrogations per orbit.
12% of Time-Missions I & II.
3 6 minute interrogations per 24 hours for Mission III.

Qualification: Unit qualified for use on Sert II and on Isis.

Availability: Will be available for fall, 1973.

The unit consists of 2 decoders in one package, each having 216 commands capability.

ITEM 20 - PROGRAMMER CLOCK (All missions)

The unit consists of an 107.52 KHz oscillator and digital countdown circuitry and performs 4 functions:

- 1) Data encoder sequencing signals providing frame, word, and bit control.
- 2) Time signals to the experiment of 10 minute and 1 hour intervals.
- 3) Spacecraft control pulses for stabilization timing and other automatically timed functions.
- 4) Clock count accumulation for time tagging the stored and real-time data samples.

Duty Cycle: Continuous Operation (100%)

Input: 130 Milliwatts

A new fabricated item.

CLOCK OUTPUT

Data Encoder	Major frame rate	1/minute
	Minor frame rate	2/sec.
	Word rate	240/sec
	Bit rate	1680/sec.
	Sequence logic for encoder	
Experiment	Ten minute control pulse	
	Hour pulses (48 discrete codes)	
Accumulator	Half minute pulses	
	Accumulator output	
Spacecraft Timed Controls	Transmitter Auto off (11 Min. lapse)	
	Stabilization time intervals	

ITEM 21 SIGNAL CONDITIONER - All Missions

The conditioner will standardize all non-standard data voltages to a compatible level with the encoder. Also, isolation of data signals will be provided to prevent external malfunctions from affecting the primary subsystem operations. Sensor excitation will be provided within this unit.

Input: 1.0 watt at 15 VDC.

Duty Cycle: Compatible with Encoder - 15%

A new fabricated item.

ITEM 22 ENCODER

1 unit on each Bioresearch Module
SCI Electronics, Inc., Model 680 PCM T/M system

Input: Missions I & II 8.16 watts max. @ 28 ± 4 VDC
Mission III 7.36 watts max. @ 28 ± 4 VDC

Duty Cycle: Missions I & II 15% (14.4 min./orbit)
Mission III 15% (3.6 Hrs/24 Hrs.)

Qualification: This unit is a second generation and the preceding unit (Model 650) was qualified by Sandia.

Availability: Will be available for Fall, 1973

Remarks: Sandia used Model 650 on many flights;
EMR used 650 on Army AIDAS Program;
NASA Edwards has eight 680 systems.

ITEM 23 T/M TRANSMITTER AND BEACON - Missions I & II

2 T/M transmitter-beacons on module
SCI P/N 1510100-1, SCI Electronics, Inc., Off-the-shelf.

Input: 25 ma at 28 VDC (700 milliwatts)

Output: 250 milliwatts, 136-137 MHz, .002% stability

Duty Cycle: One transmitter will serve as beacon at 136 MHz and
as reserve T/M transmitter and will operate continuously
(100%)

The other will serve as T/M transmitter at 137 MHz and

as reserve beacon. As T/M transmitter it will have a 12% duty cycle (two interrogations per orbit).

If either fails, the other will perform both duties.

Qualification: By University of Iowa, 2nd quarter 1967.

Availability: Will be available for Fall, 1973

ITEM 24 DATA STORAGE

1 unit on each Bioresearch Module
Electronic Memories, Model SEMS-5L, Off-the-shelf.

Input:	0.15 watts standby	Voltages: +15 +5% UDC
	1.25 watts operate	+5 \pm 5% UDC
		-5 \pm 2% UDC

Any sequence.

Duty Cycle: 15% of time (14.4 min/orbit) Missions I & II
15% of Time (3.6 Hrs/Day) Mission III

Qualification: The SEMS-5L is the low standby power version of the SEMS-5 used on RMS.

Availability: Will be available for Fall, 1973.

ITEM 40 INTEGRATING RATE GYROS - Mission I & I(S)

2 units on each Mission I and I(S)
Honeywell P/N GG1101, Off-the-shelf.

Input: 3 watts at 28 VDC

Duty Cycle: As the spacecraft enters the dark side of an orbit, the gyros are turned on to maintain vehicle attitude. (37.3% max. in one orbit).

Qualification: Similar to GG 1111 used on SAM-D and astronaut maneuvering unit.

Availability: Will be available for Fall, 1973.

ITEM 43 EXTENDIBLE BOOMS - Mission II

3 Units are on each Mission II

Spar Model A-18, Spar Aerospace Products, Ltd., Off-the-shelf.

Input: 4 watts at 28 VDC includes DC-AC conversion and AC servo motor.

Duty Cycle: Less than 1%. Retraction or extension of the booms is a function of desired G-level and experiment.

Qualification: The A-18 is qualified for space flight and environment and has flown on Scout, Delta, Titan IIIC, Rubis, and Atlas Agena. It has been used on GGTS, GGII, 060 "A" and 'B', LISOS, ARSP, EOLE, and Probe I.

Availability: Will be available for Fall, 1973.

ITEM 44 HORIZON CROSSING INDICATORS - Missions II & III

2 units are used on each spinning mission.

Barnes P/N 13-206, Off-the-shelf with modification to interface with Module spin rate.

Input: 160 milliwatts at 28 VDC

Duty Cycle: The indicators will be employed during the first orbital pass through the ecliptic and approximately 3 months later to direct cold plate away from earth.

Qualification: Same (except power supply) as model 13-205SC used on TIROS.

Availability: Will be available for Fall, 1973.

ITEM 47 GRARR TRANSPONDER - Mission III

2 units on each Mission III vehicle.

Input: 1.61 watts +12 VDC R&RR & Command
 16.5 watts at 28 VDC transmitter

Output: 7 watts at 136 MHz

Duty Cycle: 3 Six-minute Interrogations per 24 hours
 (1.25% of time)

Qualification: Designed to meet requirements of user spacecraft.

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REFERENCES

- A-1 "Study and Analysis of Satellite Power Systems Configurations for Maximum Utilization of Power", NASA Report CR-898, October 1967.
- A-2 "Bioexplorer Spacecraft Preliminary Design Study", Contract NAS 2-6028, LTV/VMSC Report T146-1, Vol. I -Basic Report, Vol. II - Appendices, 29 November 1970.
- A-3 J. Burnett, S. Friedlander, M. Klein, "Selected Notes on Solar Power Systems", EOS Report, 7 July 1967 .
- A-4 E. L. Ralph, "Performance of Very Thin Silicon Solar Cells," Heliotek Report, March 1967.
- A-5 E. L. Ralph, J. H. Martin, "Radiation Damage to Thin Solar Cells," Heliotek Report August 1967.
- A-6 "Modified Experiment Support Requirements to be Evaluated Under Task I-1", Enclosure (1) to NASA/Ames letter PEF: 204-5 (80L) dated 11 June 1971.

APPENDIX B
REVISED THERMAL PROFILE ANALYSIS

1.0 INTRODUCTION

The thermal analysis presented in this appendix was performed to determine the impact on the thermal control system design of the revised heat loads and performance requirements presented in Reference B-1. These revised requirements for Missions I, II, and III are summarized in Table 1. Also included in Table 1 for comparison are the original requirements on which the previous design and performance analysis were based (Reference B-2).

TABLE 1. - DESIGN REQUIREMENTS

	Baseline (Ref. B-2)	Revised (Ref. B-1)
Continuous Heat Load	180-270 Btu/hr	136 Btu/hr
Peak Heat Load	350 Btu/hr for 10 min/hr	408 Btu/hr for 3 min/hr
Set-Point Temperature Range	35°F - 45°F	35°F - 50°F
Temperature Tolerances	Set Point ^{+0°F} -5°F	Set Point ^{+0°F} -5°F

The thermal control system concept has been described in detail in Reference B-2. Figure B-1 is a schematic showing the basic elements of the system. A direct-radiating cold plate with a radiating area of 6 square feet is mounted on the top of the experiment package, and heat load control is provided by a two-position louver array which is positioned in either the "open" or "closed" position by the louver control unit on the basis of signals from temperature sensors at the cold plate. The temperatures at which the "open" and "close" commands occur can be varied to achieve the desired set-point range indicated in Table 1. Further control flexibility can be obtained by adjusting the louvers so that they do not fully close in the "closed" position.

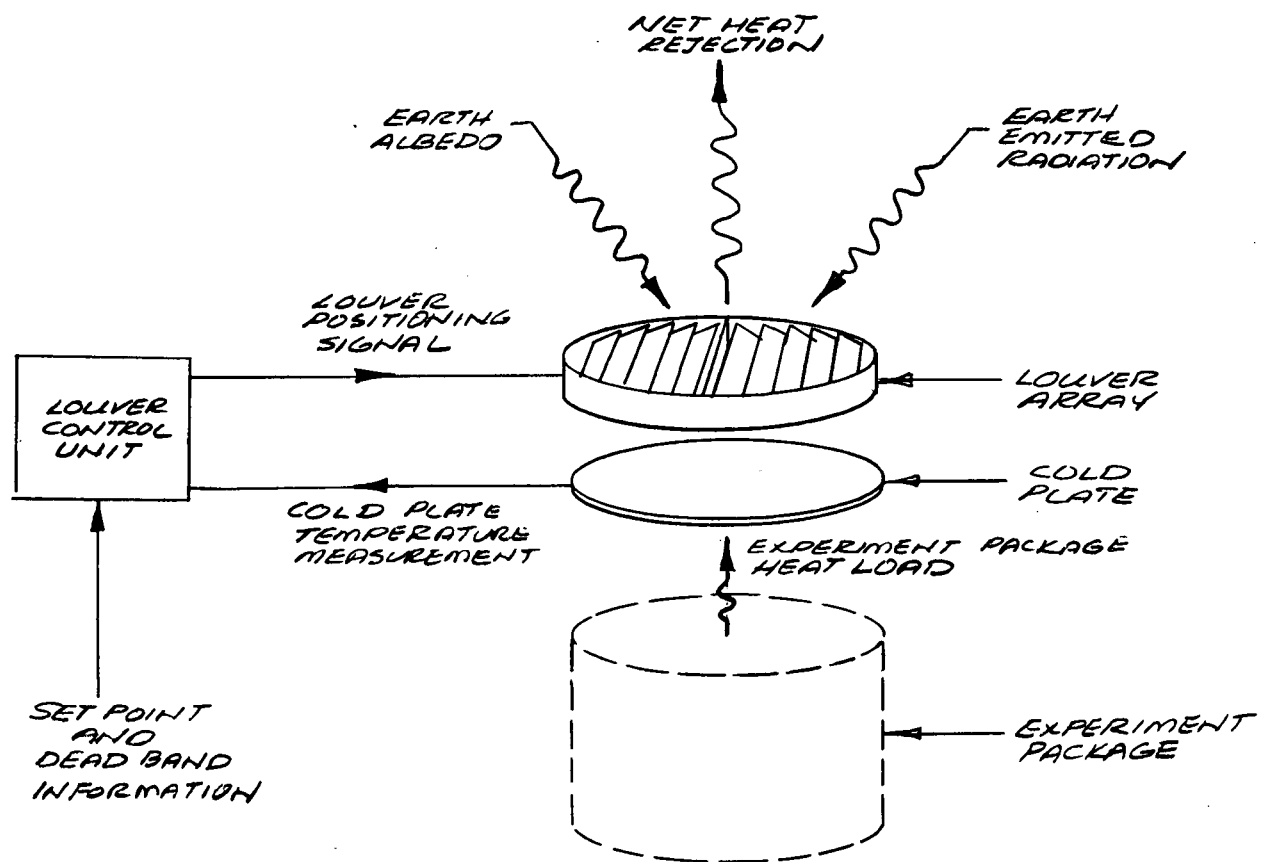


FIGURE B-1. - THERMAL CONTROL SYSTEM SCHEMATIC

2.0 THERMAL ANALYSIS

The revised thermal analysis was performed in two parts. In the first part the baseline thermal control system design (from Reference B-2) was analyzed to determine its performance with the revised heat loads and performance requirements. In the second part a parametric analysis was performed to determine the optimum design parameters for the system based on the revised requirements to determine what cost and/or weight savings were possible. The two basic design parameters considered in this analysis were cold plate mass and material, effective emissivity ratio (between the "louver open" and "louver closed" positions) and louver dead-band (temperature difference between "open" and "close" commands).

The most severe combination of internal heat load and environmental heating are presented by Mission II. This results from the fact that the Mission II spacecraft is inertially stabilized with its axis normal to the sun line, and must be facing the earth for an extended portion of each orbit. The Mission I spacecraft is assumed to be rotating in yaw at a rate of 10 revolutions per hour (one degree per second) and the earth radiation and albedo heat loads are distributed more uniformly over the orbit. Because Mission II is the most severe case, the required radiator thermal mass (or heat capacity) was established for this mission and used as the baseline for Missions I and III. The louver parameters (emissivity ratio and dead band), which are readily adjustable for each mission, were then optimized for Missions I and III. A separate analysis was performed for Mission I(S), which involves a different set of heat loads and cold plate design constraints.

A specialized computer routine, as discussed in Reference B-2, was used to perform the parametric analysis. In this routine the cold plate is considered as a single thermal node, and both internal and environmental heat loads are input as a function of time and orientation. Ten orbits were calculated for each run to assure that equilibrium conditions were obtained.

2.1 MISSION II

2.1.1 Heat Loads. - The internal (experimental) and environment heat loads assumed for the Mission II analysis are shown in Figure B-2. The environmental loads are identical to those used in Reference B-2, and are based on the most severe levels that would occur at a solstice condition for a vehicle launched at a 37-degree inclination. It has been assumed that the spacecraft can be rotated 180 degrees once during a six-month mission so that the cold plate will be facing the earth only on the dark or "night" side and solar albedo effects are minimized. Peak internal heat loads of 3-minute duration were included once an hour, with one peak occurring at the time of maximum environmental heating.

2.1.2 Baseline Design Performance. - The "baseline" design is defined in all cases as that which was presented in Reference B-2. For Mission II

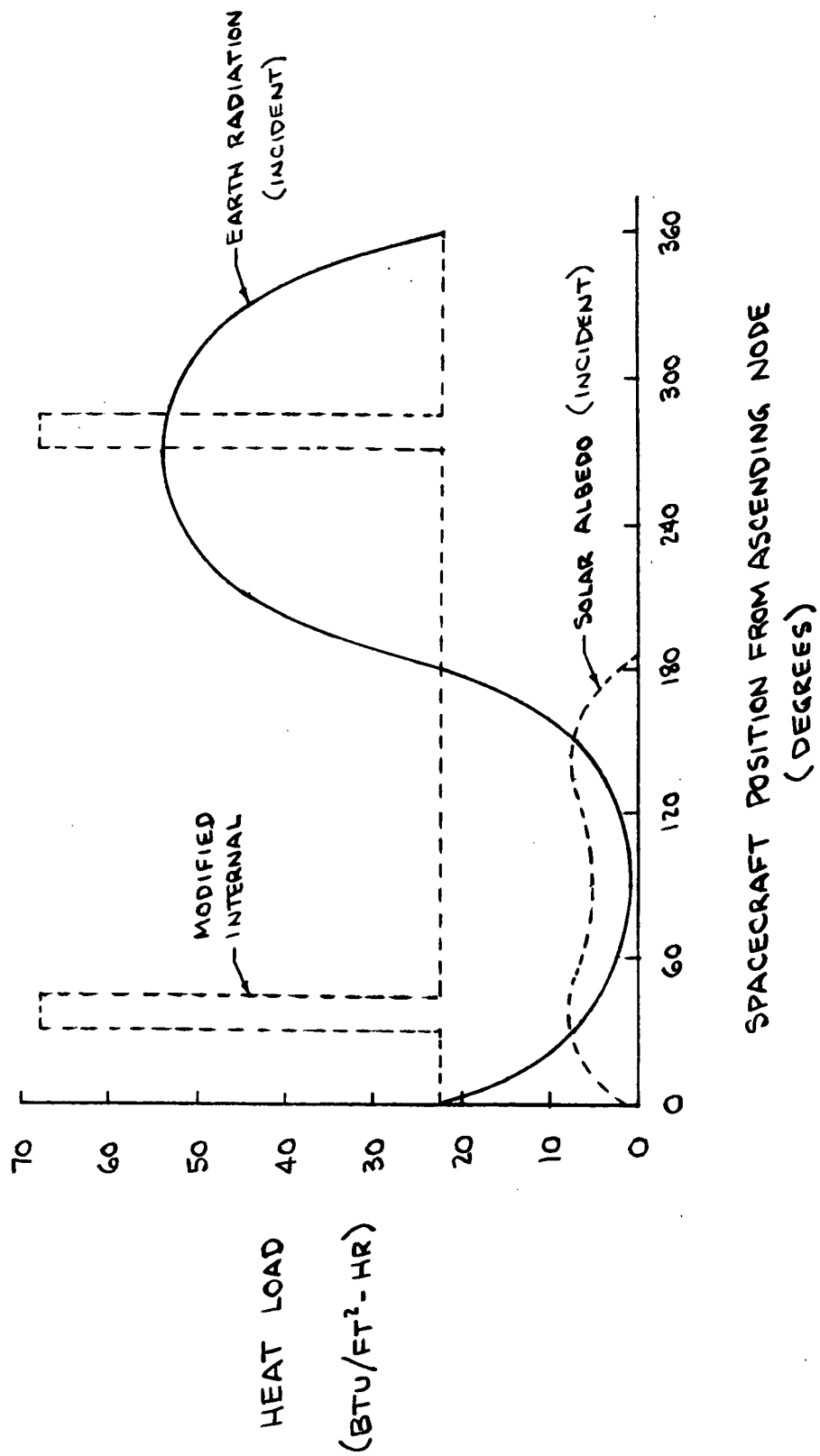


FIGURE B-2. - MISSION II HEAT LOADS

the baseline design was defined by the following parameters:

Cold plate material - beryllium
Cold plate mass - 23 pounds
Emissivity ratio - 3.0
Louver dead-band - 2.5°F

The performance of this baseline design for the revised heat loads is shown in Figure B-3. Cold plate temperature is shown as a function of spacecraft position over a characteristic orbit. Performance at the minimum set-point of 35°F is shown, since this represents the most severe case. The baseline design meets the revised heat load requirements, with a maximum louver cycling rate of approximately 1.0 cycle per hour or 1.5 cycles per orbit.

2.1.3 Revised Design Analysis. - The assumptions made in the design analysis and the manner in which the analysis was performed are illustrated by a sample run shown in Figure B-4. Because the continuous heat load is lower for the revised requirements, the cold plate temperature may decrease during low environmental heating conditions, even with the louvers closed. This decrease is shown by (A) in Figure B-4. This decrease is a function both of radiator thermal mass and of effective emissivity at the closed position. The temperature for closing the louvers must be set at least the number of degrees indicated by (A) above the lower temperature limit to assure that the temperature of the cold plate will never drop below the limit.

Similarly, when peak internal heat loads occur at the most severe environmental conditions, the cold plate is not capable of rejecting the heat with the louvers fully open. The cold-plate temperature rise for this condition is illustrated by (B) in Figure B-4. The louver-open setting must be at least this amount below the upper temperature limit. The difference between the upper and lower limits then defines the maximum allowable deadband for a given cold plate mass and louver emissivity ratio. This allowable deadband is also affected by the required range of set-point temperatures, since the lower limit is determined by conditions at the maximum set point (50°F), where the temperature decrease at minimum heat load is greatest, and the upper limit is determined by the minimum (35°F) set point.

Figure B-5 shows the allowable dead-band as a function of both radiator (or cold-plate) mass and emissivity ratio. Cold-plate mass is plotted in units of pounds of aluminum, but this designation is arbitrary. The analysis was performed parametrically in terms of heat capacity in Btu/°F. Corresponding values for a beryllium cold-plate can be obtained by assuming that the required mass is approximately one-half that for aluminum.

In addition to minimizing required cold-plate mass, to reduce weight and cost, it is desirable for reliability to minimize the cycling rate of the louvers. Louver cycling rates as a function of cold-

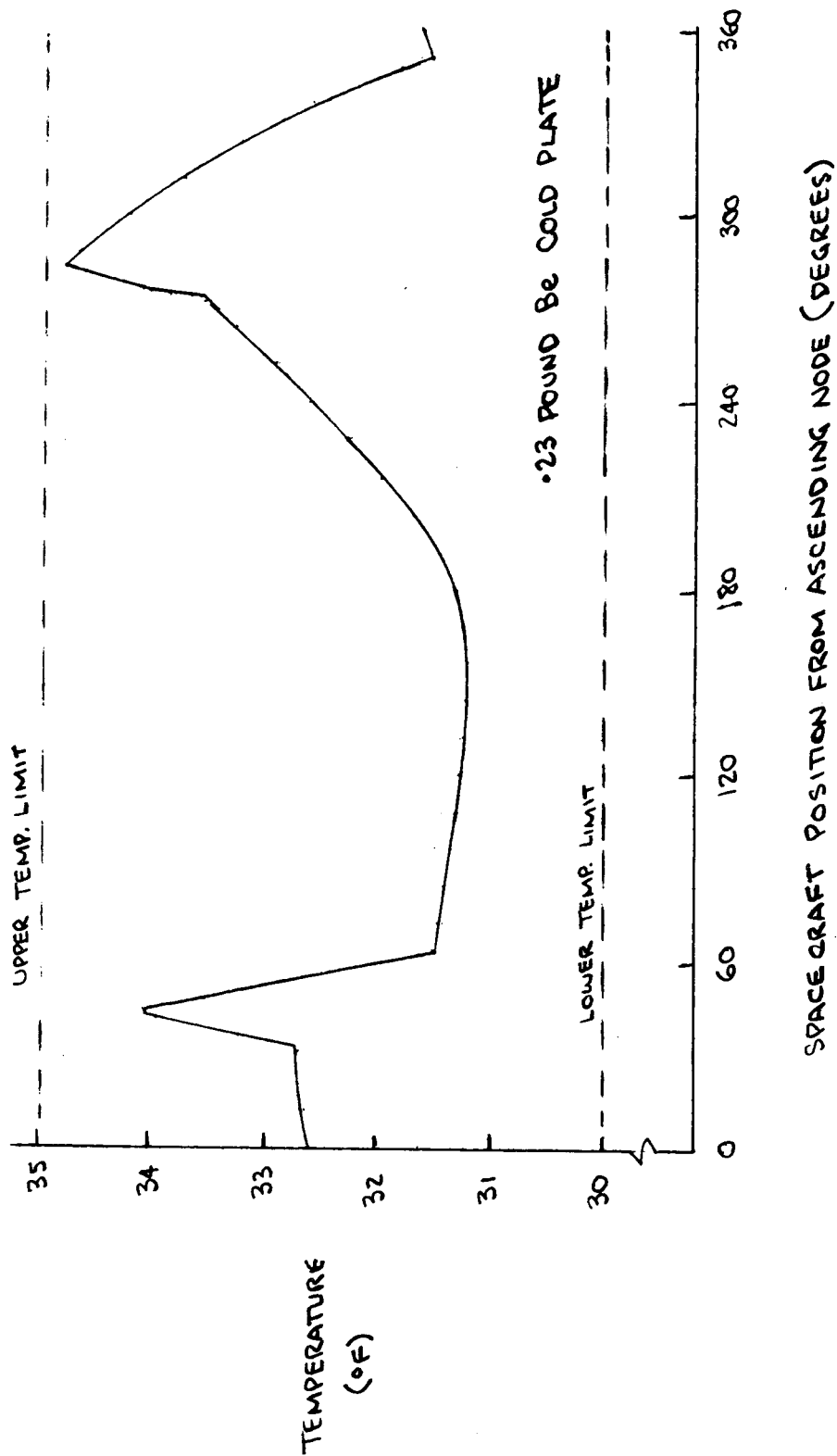


FIGURE B-3. - BASELINE COLD PLATE TEMPERATURE PROFILE - MISSION II

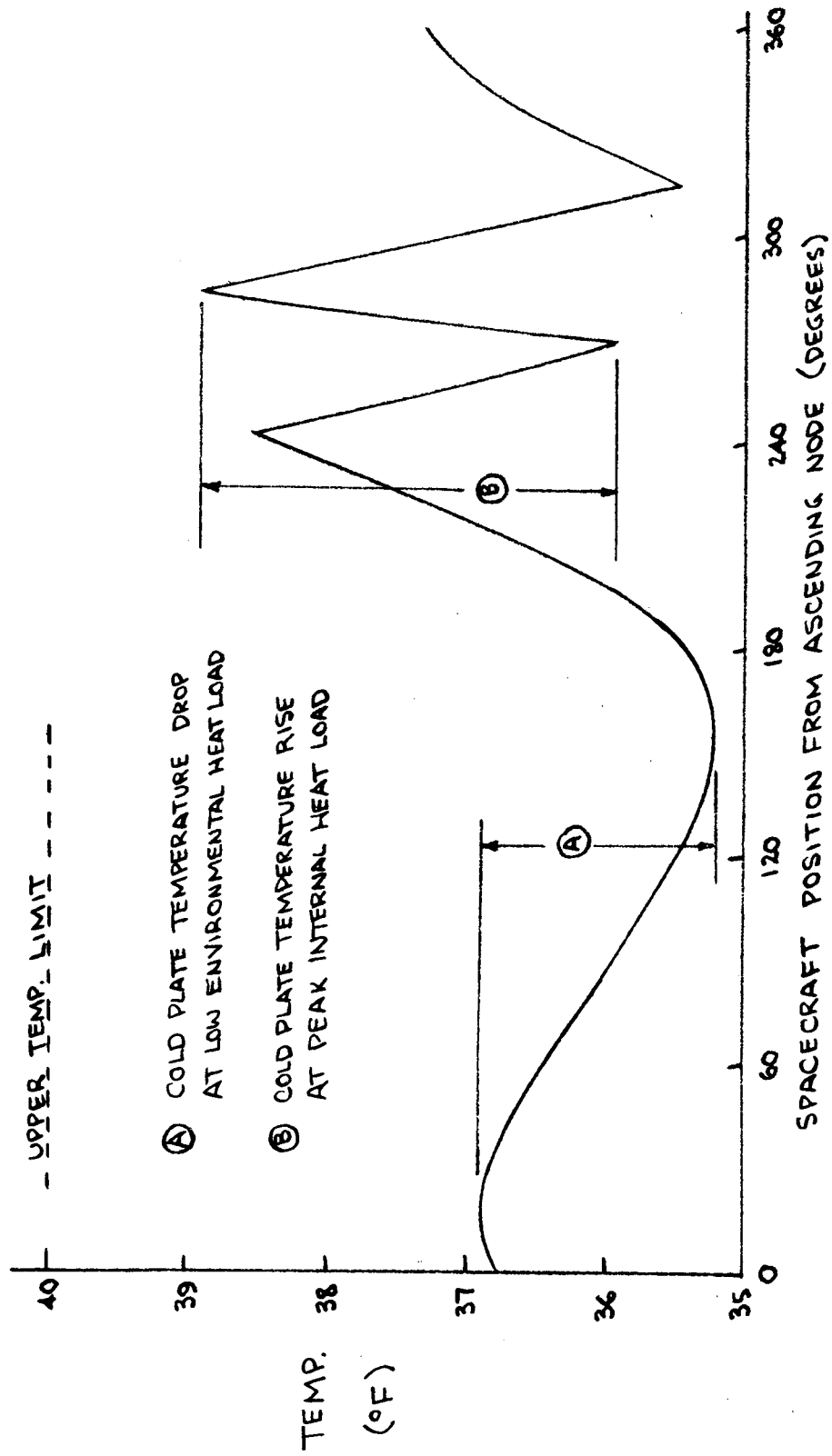


FIGURE B-4. - TYPICAL COLD PLATE PERFORMANCE - MISSION II

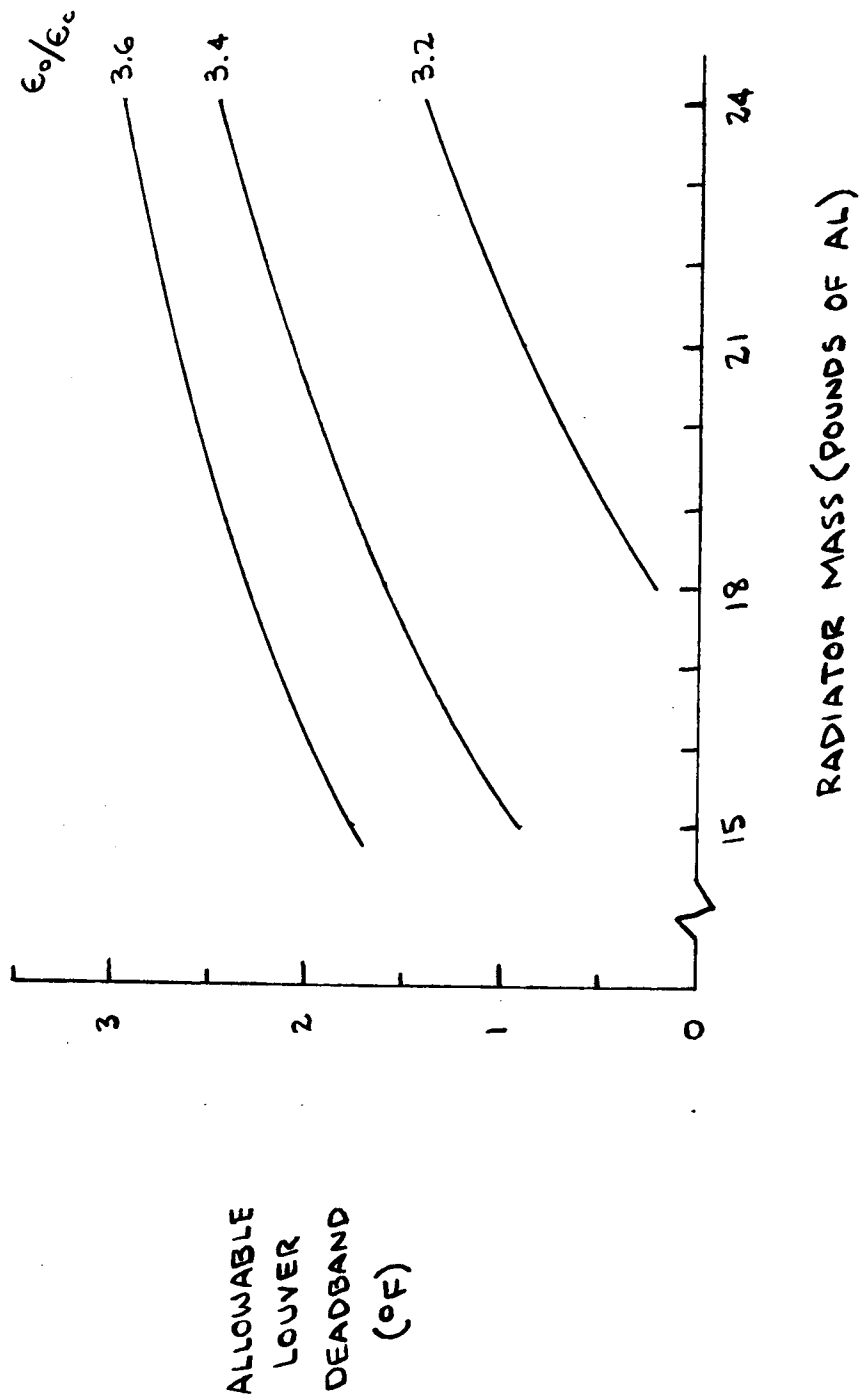


FIGURE B-5. - ALLOWABLE LOUVER DEADBAND - MISSION II

plate mass are shown in Figure B-6 for several values of emissivity ratio, with dead-band limitations shown in Figure B-5 implicit in the analysis. The relatively narrow range of emissivity ratios shown result from the fact that for reasonable values of cold plate mass a ratio of 3.2 is the lowest value that permits any dead-band at all, and at a ratio of 3.6 the temperature drop at minimum heat load (A in Figure B-4) is negligible and no further advantage can be gained by increasing the ratio.

Figure B-6 shows that an emissivity ratio of 3.6 is optimum for cold-plate masses up to approximately 22 pounds, and this ratio was selected for the revised design. The expected range in louver cycling rate for various set-point temperatures and sequencing of the peak internal heat load (in terms of peaks per hour) is shown in Figure B-7. A cold-plate mass of 21 pounds of aluminum was selected as the design point, since the louver cycling rate becomes relatively flat at this point. The range in louver cycling rate is approximately 3-5 cycles per orbit, slightly higher than was obtained for the baseline design in Reference B-2.

2.1.4 Revised Design and Performance. - The design parameters for the thermal control system for the revised heat load requirements are:

Cold-plate material - aluminum
Cold-plate mass - 21 pounds
Emissivity ratio - 3.6
Louver dead-band - 2.6°F

The temperature profiles for this design over two spacecraft orbits at the minimum set-point temperature are shown in Figure B-8. Also included for reference is louver position. One significant factor in Figure B-8 is that the cold-plate temperature over the two orbits remains in a narrow 3-degree band near the lower limit of the allowable temperature range. This indicates that, if very short and infrequent temperature transients which exceed the upper temperature limit were permitted, the louver dead-band could be opened up, with a consequent reduction in louver cycling rate.

2.2 MISSION I

2.2.1 Heat Loads. - The internal heat loads and incident earth radiation and solar albedo for Mission I are shown in Figure B-9 as a function of spacecraft orbital position. As in Mission II, the incident heat curves are based on a solstice condition where the spacecraft is at a maximum orbital inclination with respect to the equator and the consequent view angle between the cold plate and the earth is also maximum. The cyclic shape of the incident heat curves results in the 10 revolution/hour yaw rate of the spacecraft.

2.2.2 Baseline Design Performance. - The "baseline" design for Mission I was defined by the following parameters:

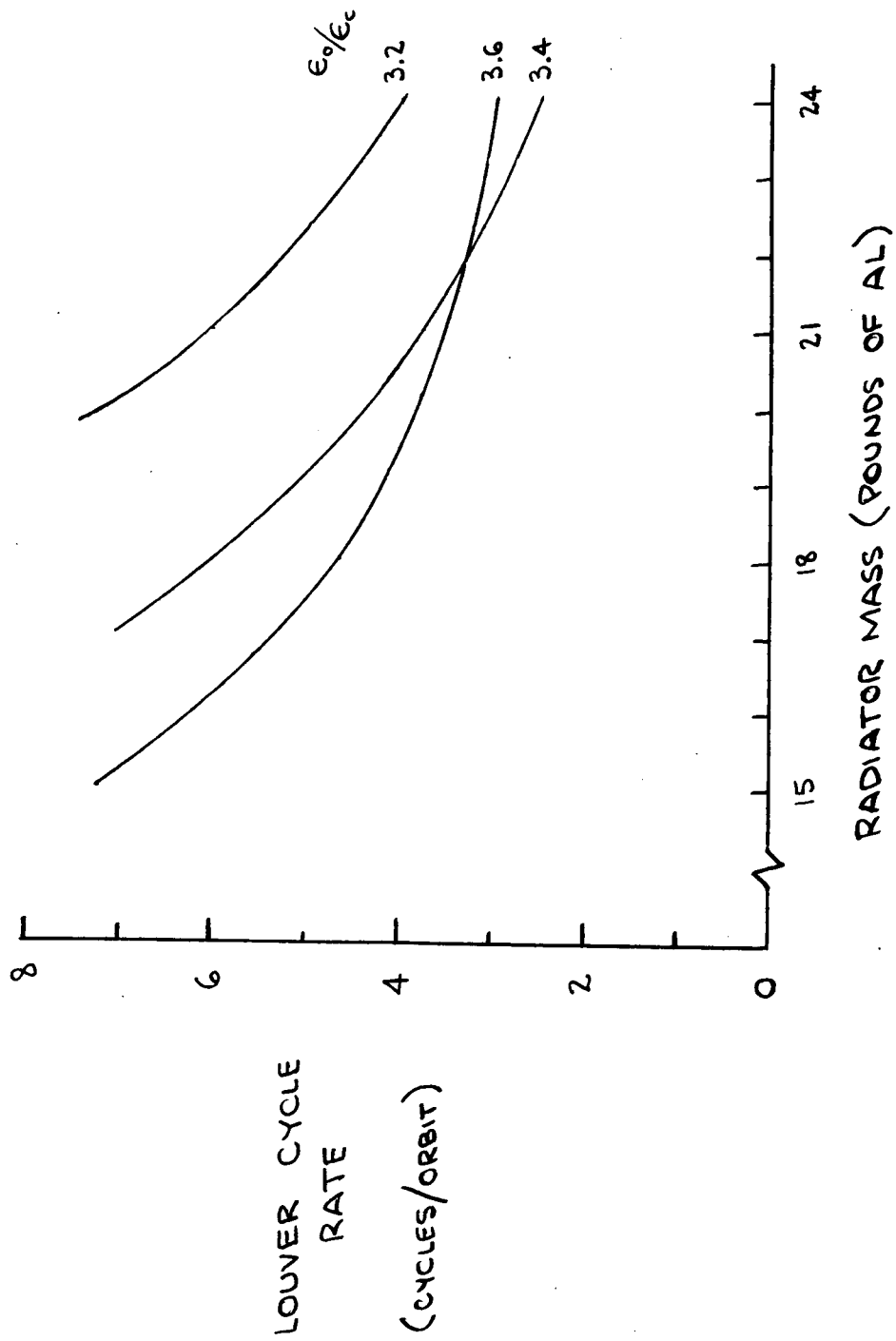


FIGURE B-6. - LOUVER CYCLING RATE - MISSION II

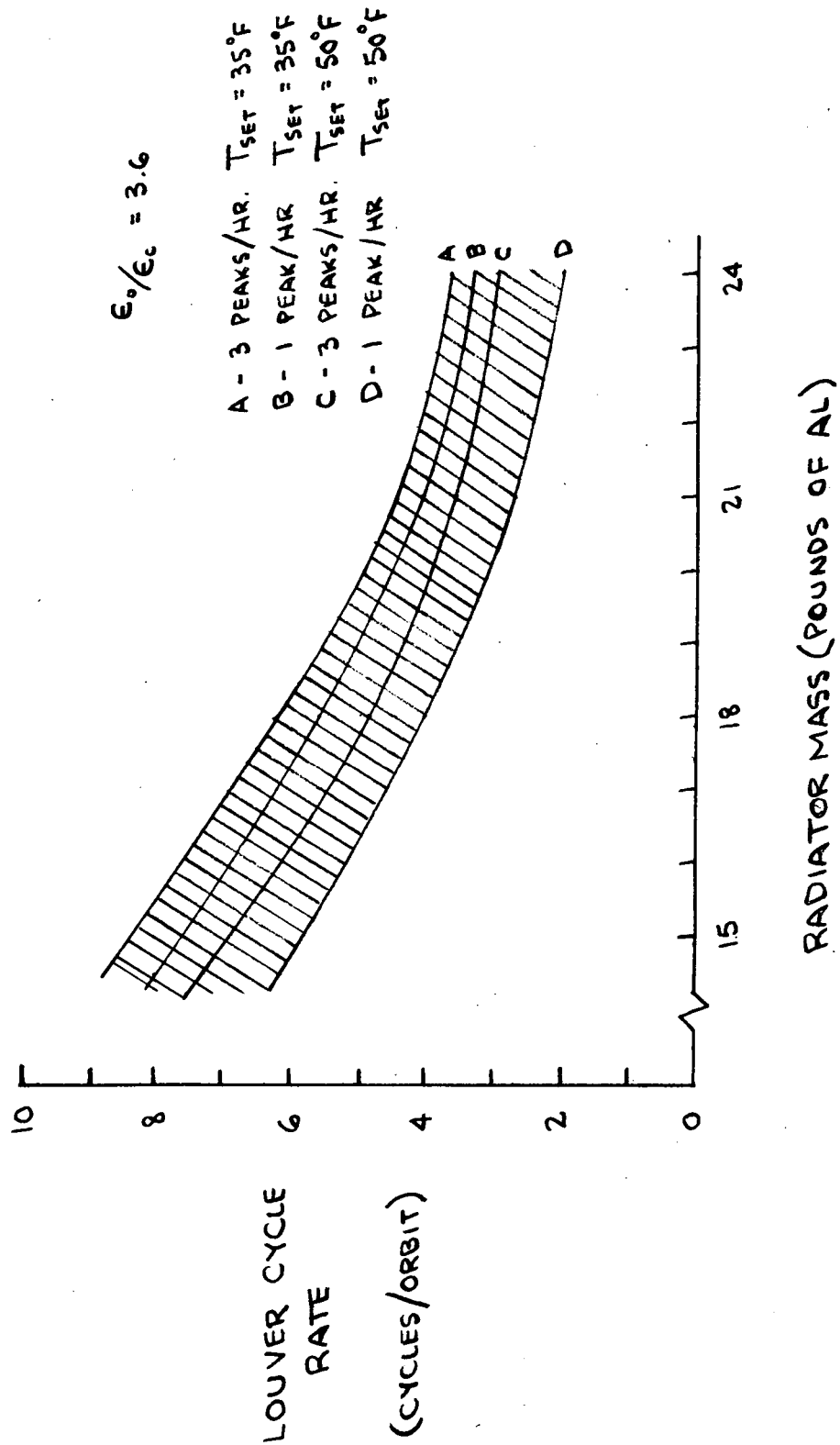


FIGURE B-7. - LOUVER PERFORMANCE RANGE - MISSION II

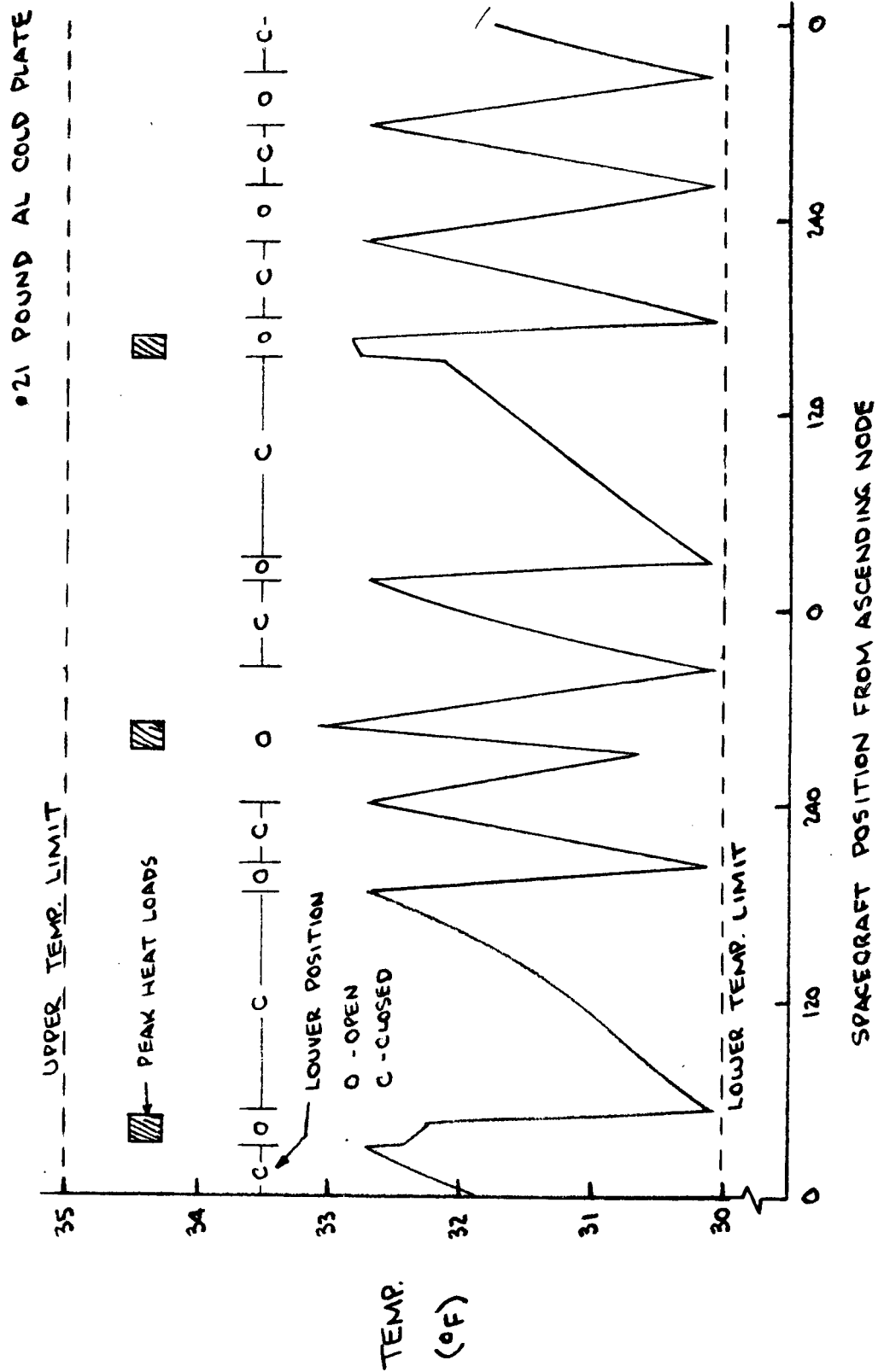


FIGURE B-8. - COLD PLATE TEMPERATURE PROFILE - MISSION II

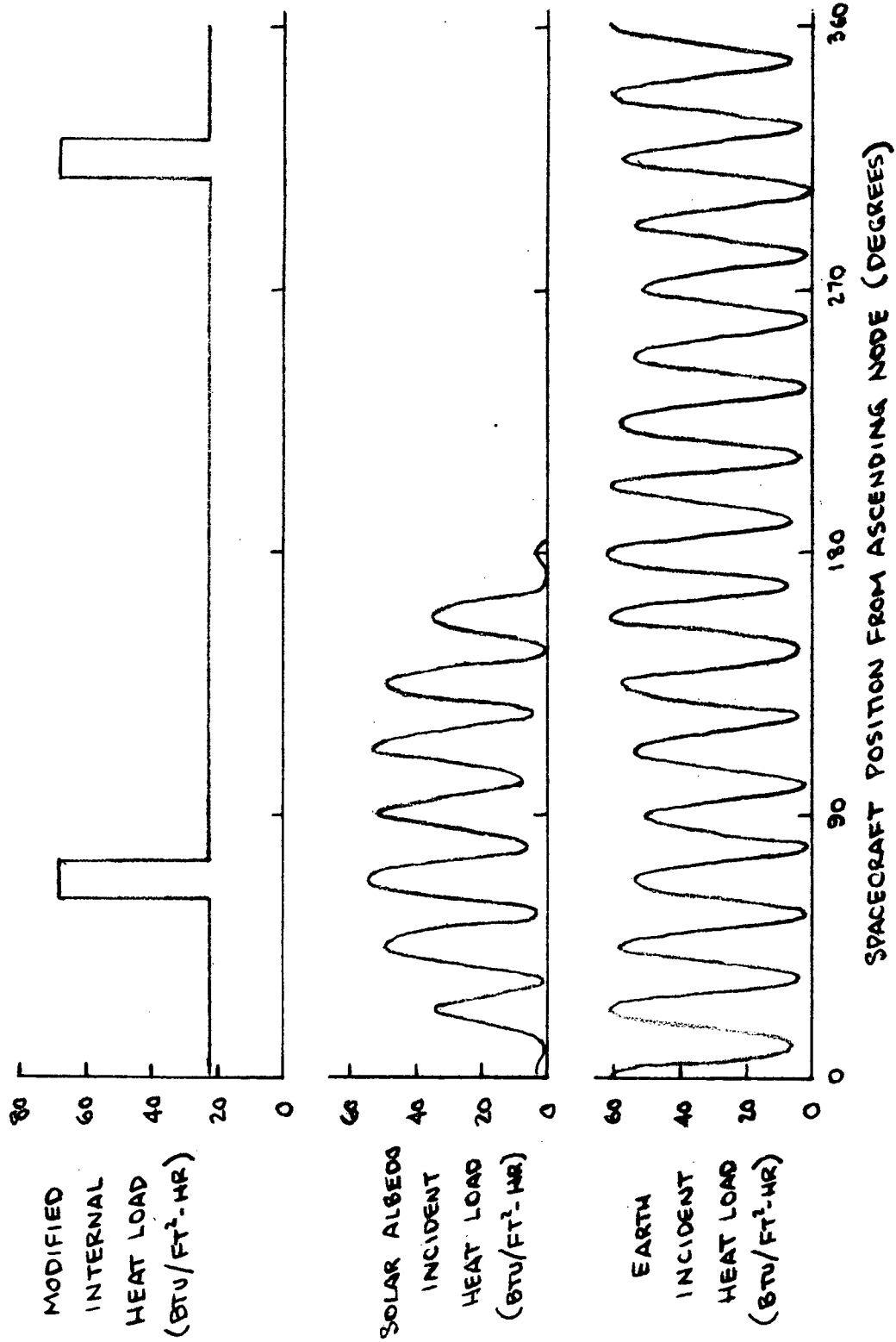


FIGURE B-9. - MISSION I HEAT LOADS

Cold plate material - beryllium
Cold plate mass - 23 pounds
Emissivity ratio - 3.0
Louver dead-band - 3.0°F

The performance of the baseline system with the revised internal heat loads is shown in Figure B-10. The system meets the revised loads with a louver cycling rate of approximately 1 cycle per hour.

2.2.3 Revised Design Analysis. - The cold-plate mass of 21 pounds of aluminum determined by the Mission II analysis was also used for Mission I to satisfy the requirement that the same cold plate design be applicable to all missions.

The effect of emissivity ratio on allowable deadband and louver cycling rate are shown in Figure B-11. The somewhat lower emissivity ratios for Mission I result from the fact that the yawing of the spacecraft reduces the effective orbital variation in incident heat loads from those encountered in Mission II. On the basis of this analysis, an emissivity ratio of 2.8 was selected for Mission I.

2.2.4 Revised Design and Performance. - The thermal control system design parameters for the revised heat load requirements are:

Cold plate material - aluminum
Cold plate mass - 21 pounds
Emissivity ratio - 2.8
Louver dead-band - 2.3°F

The cold-plate temperature profile for this design for a characteristic orbit and at the minimum set-point temperature is shown in Figure B-12. As is the case with Mission II, the temperatures tend to remain within approximately a 3-degree band. A reduction in louver cycling rate could be realized by opening up the dead band and permitting occasional short transients outside the 5-degree temperature band.

2.3 MISSION III

A cursory analysis was performed on Mission III to determine the performance of the baseline design and establish the limiting values for emissivity ratio and deadband for a 21-pound aluminum radiator. Since there is effectively no incident heat load except for very short periods near perigee, the analysis consists of determining the minimum allowable emissivity ratio that will assure that the cold plate temperature will not drop below the minimum temperature limit at the continuous heat load, and the maximum deadband that will assure the cold plate temperature does not exceed the maximum temperature limit at peak heat load.

The Mission III "baseline" thermal control parameters are:

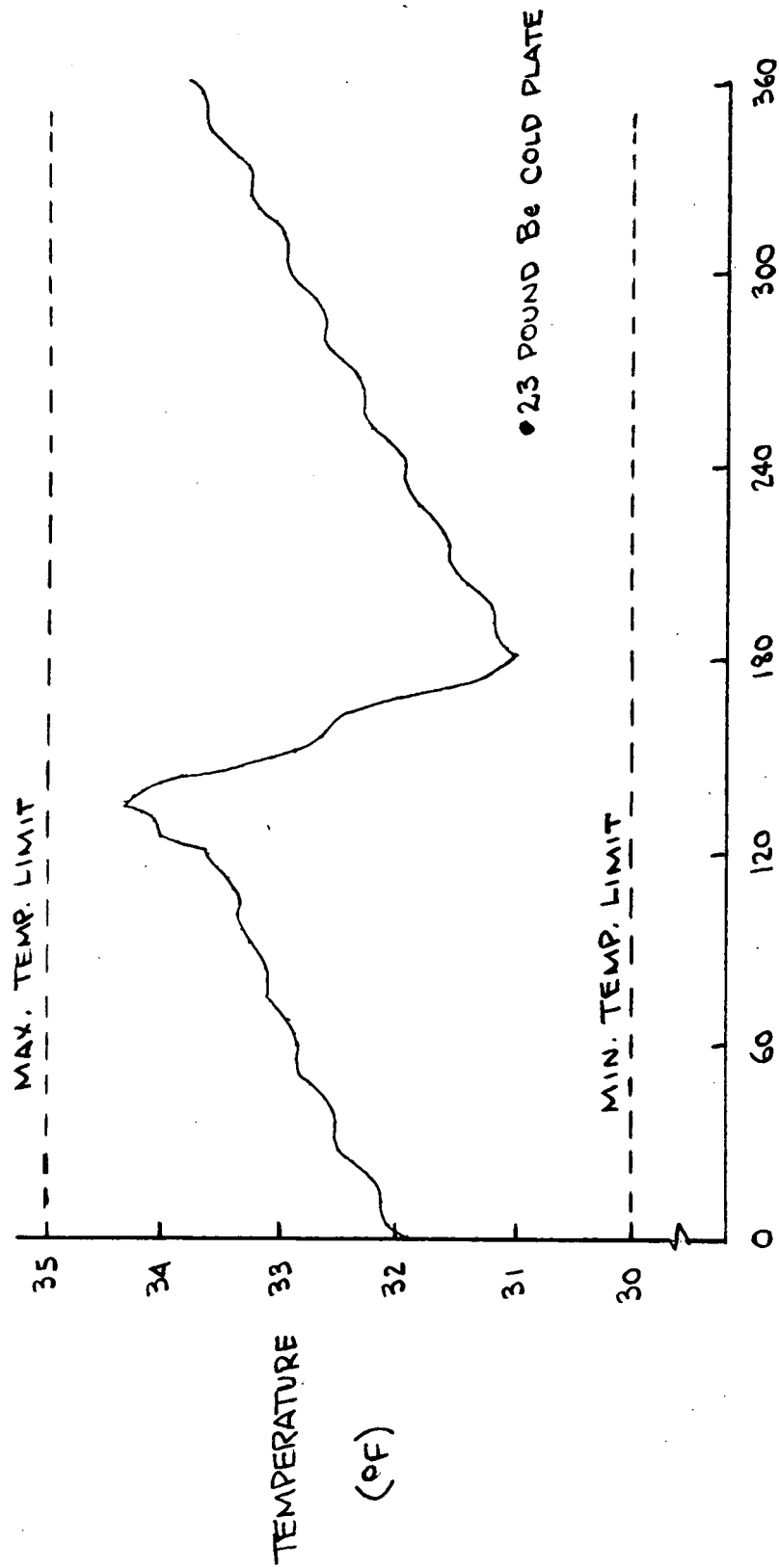


FIGURE B-10. - BASELINE COLD PLATE TEMPERATURE PROFILE - MISSION I

C.2.

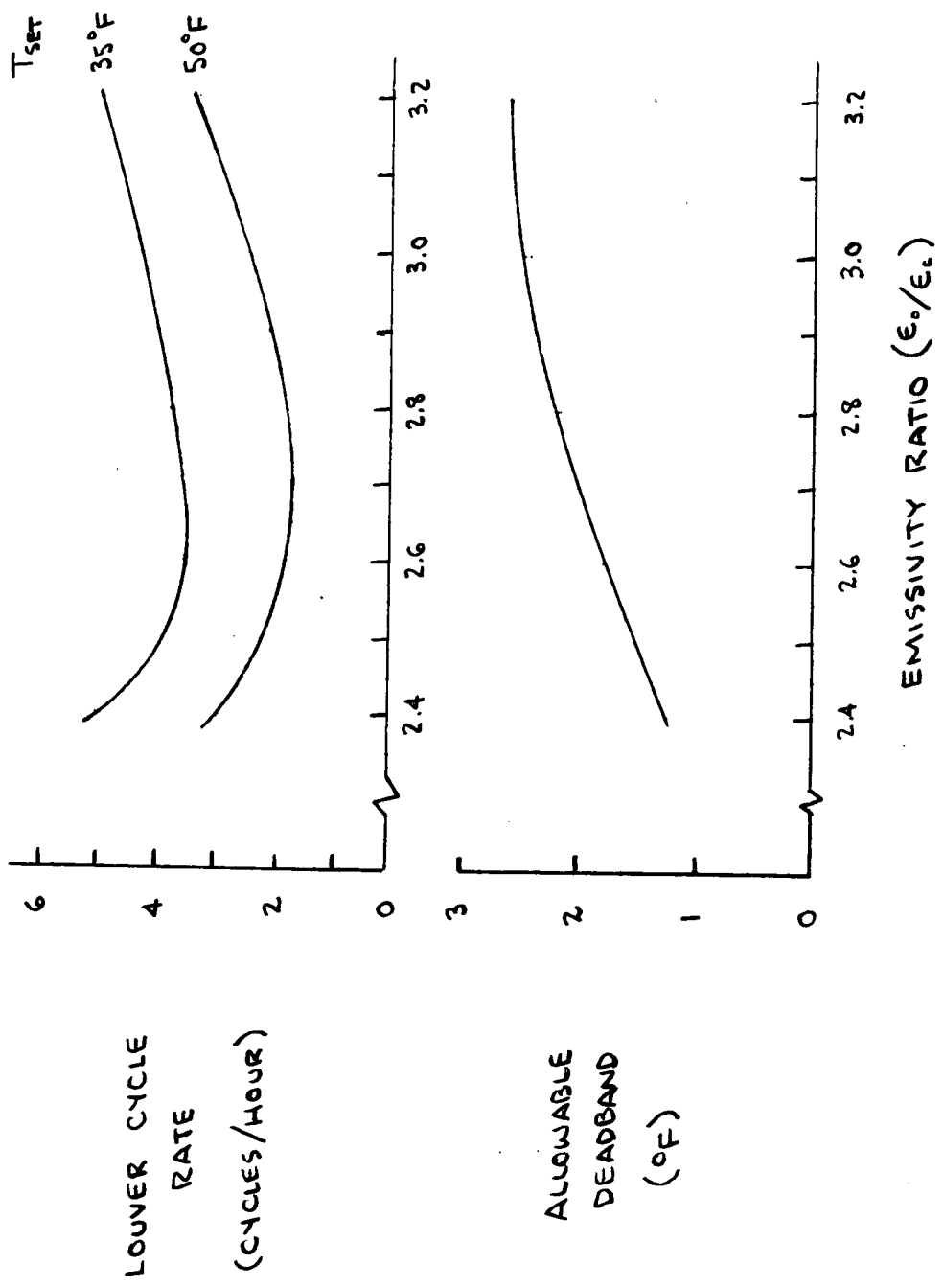


FIGURE B-11. - LOUVER PERFORMANCE - MISSION I

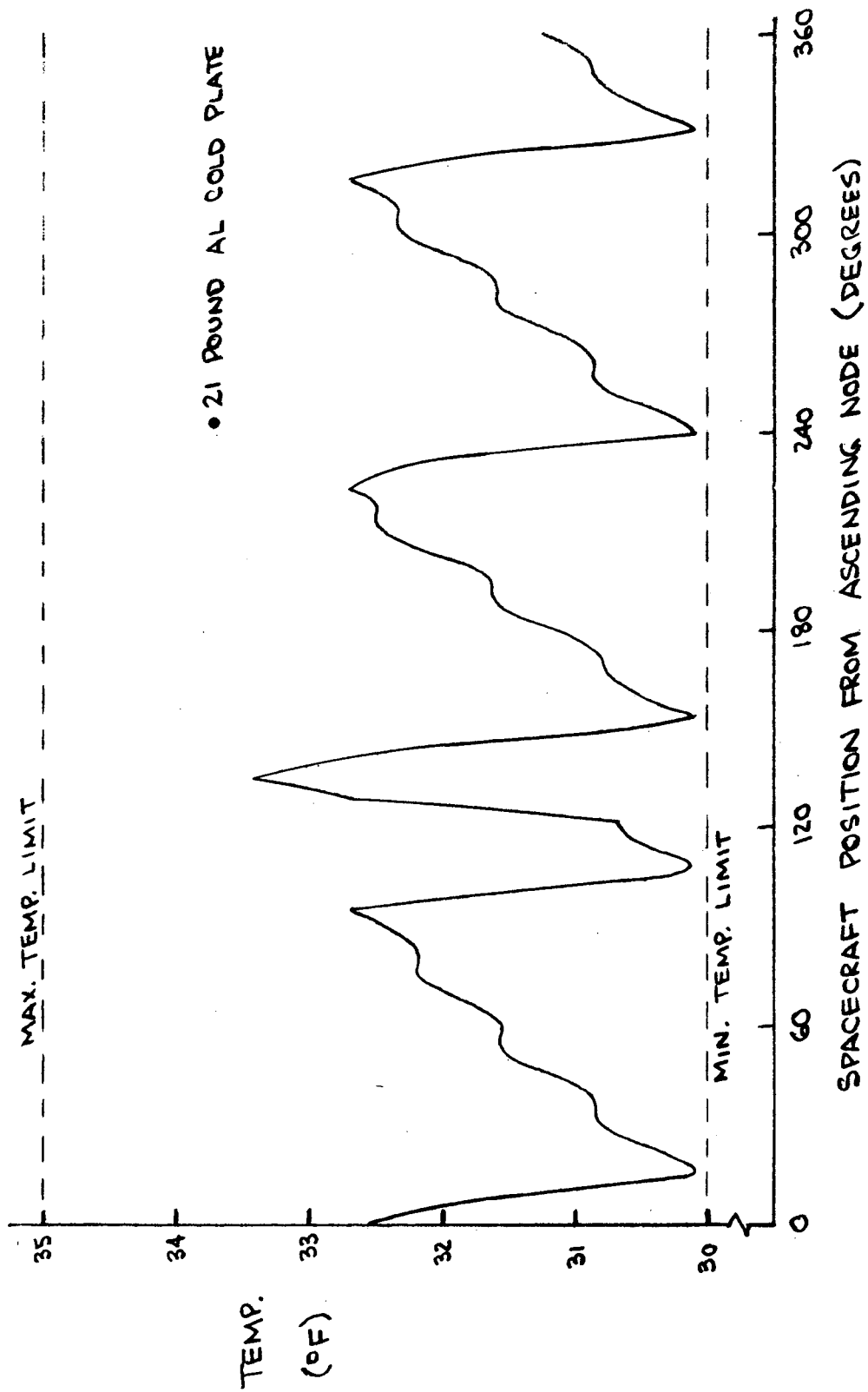


FIGURE B-12. - COLD PLATE TEMPERATURE PROFILE - MISSION I

Cold plate material - beryllium
Cold plate mass - 23 pounds
Emissivity ratio - 3.0
Louver dead-band - 3.0°F

Because of the lower continuous heat load in the revised requirements, the baseline design will not meet the requirements without an increase in emissivity ratio to a minimum of 3.6. For a 21-pound aluminum cold plate the design parameters are:

Cold plate material - aluminum
Cold plate mass - 21 pounds
Emissivity ratio - 3.8
Louver dead-band - 4.0°F

The louver cycling rate for both the baseline design and the revised design is less than 1 cycle/hour.

2.4 MISSION I(S)

Mission I(S) represents a special case, since the change in heat load requirements permits a change from a pumped fluid system with side-mounted radiators, as described in Reference B-2, to a direct-radiating cold plate mounted on the top of the experiment package. The performance requirements for the Reference B-2 study were identical to those listed in Table 1 for the other missions. The available cold plate area for Mission I(S) is only 2.73 square feet, however, rather than the 6.0 square feet available for the other missions, which was inadequate to reject the specified heat load. The revised heat load requirements, which are tabulated in Table 2, can be met with the smaller cold-plate radiating area.

TABLE 2. - REVISED MISSION I(S) DESIGN REQUIREMENTS

Continuous Heat Load	100 Btu/hr
Peak Heat Loads	185 Btu/hr for 2 min/orbit
	340 Btu/hr for 15 sec/orbit
Set Point Temperature Range	35°F - 50°F
Temperature Tolerances	Set Point +0°F -5°F

The incident heat loads for Mission I(S) are identical to those for Mission I as shown in Figure B-9. For the performance analysis

it was assumed that peak internal loads occurred sequentially at a point where the environmental loads were maximum. The design parameters for the thermal control system for the revised requirements are:

Cold Plate Material - aluminum
Cold Plate Mass - 12 pounds
Emissivity Ratio - 2.0
Louver Dead Band - 3.0°F

The temperature profile for this design at the minimum set point is shown in Figure B-13. The louver cycling rate is slightly less than 4 cycles/orbit. As was true for the other missions, the 3-degree range of the temperature indicates that the dead band can be increased, and the louver cycle rate reduced, if occasional short-duration temperature transients outside the 5°F tolerances are permitted.

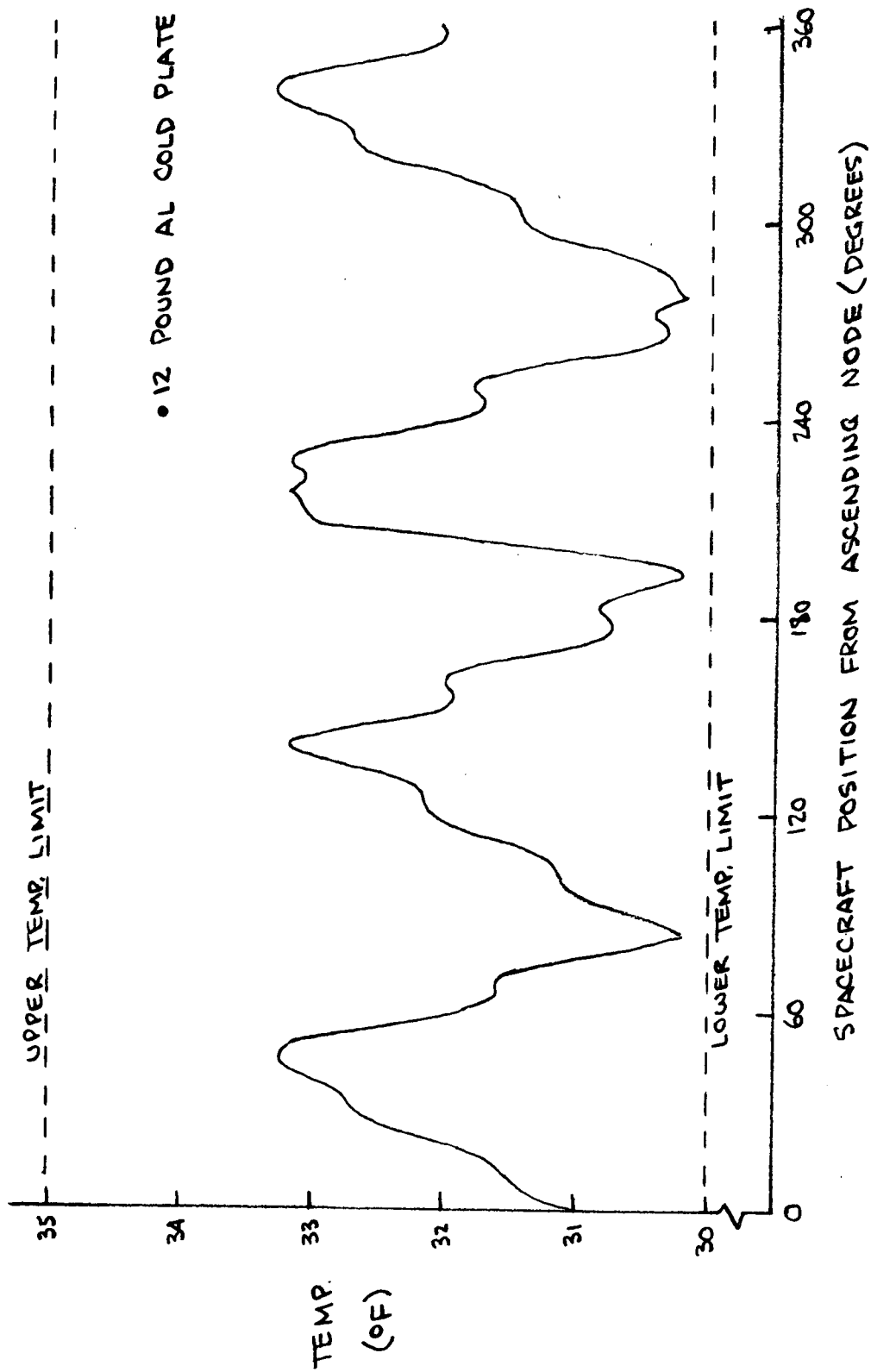


FIGURE B-13. - COLD PLATE TEMPERATURE PROFILE - MISSION I(S)

3.0 SUMMARY

The results of the revised thermal analysis show that overall the modified heat load and performance requirements present less severe design problems than the baseline requirements analyzed in Reference B-2. The following conclusions represent a summary of the thermal analysis.

- (1) The "Baseline" design (Reference B-2) will meet all modified heat load requirements for all missions.
- (2) The modified heat loads permit the use of an aluminum cold plate, or of a lighter beryllium cold plate.
- (3) The modified requirements for Mission I(S) permit the use of a direct-radiating cold plate rather than a pumped-fluid system.
- (4) The louver cycling rate is slightly higher for the revised design than for the baseline system in Reference B-2. The baseline design will meet the revised heat loads, however, with a lower louver cycling rate than with the previous heat loads.
- (5) Louver cycling rate could be reduced if occasional short-duration cold plate temperature transients outside the 5°F temperature tolerances were permitted.
- (6) The increase in set-point temperature range from 10°F to 15°F has very little effect on performance.

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REFERENCES

- B-1. "Modified Experiment Support Requirements to be Evaluated Under Task I-1," Enclosure (1) to NASA/Ames letter PEF: 204-5 (80L) dated 11 June 1971.
- B-2 LTV/VMSC Report No. T146-1 "Bioexplorer Spacecraft Preliminary Design Study," Vol. I - Basic Report, Vol. II - Appendices, 29 November 1970.

APPENDIX C

DESIGN ANALYSIS

OF

TELEVISION MONITOR

APPENDIX C

SUMMARY DESCRIPTION OF EXPERIMENT

MONITORING AND S-BAND SYSTEM

APPENDIX C

SUMMARY DESCRIPTION OF EXPERIMENT MONITORING AND S-BAND SYSTEM

1.0 INTRODUCTION

The baseline Bioresearch Module is equipped with a VHF communications system (148 MHz Uplink and 137 MHz, 30 KHz Bandwidth, Downlink) which interfaces with the STADAN tracking system. In accordance with Reference C-1, a preliminary design is defined for a television system to monitor biological activity in the experiment package. Since TV transmission will require large bandwidths, an S-Band communications system is also defined.

2.0 EXPERIMENT MONITORING

The TV system is required to operate one minute per orbit at either 24 frames/sec or one frame/minute, with a resolution of 1/10 inch in a 7.5 inch by 10 inch frame size. The information can either be real time or stored.

The minimum resolution required is 200 TV lines which can be reproduced using a 300 line scan rate. The 300 lines require a 3 MHz bandwidth which can be obtained by employing an S-Band downlink of 2300 MHz.

The Bioresearch Module VHF baseline system can neither transmit nor store the required monitoring information.

2.1 INTEGRATION WITH BIORESEARCH MODULE

A parallel TV subsystem is required for experiment monitoring. Figures C-1 and C-2 illustrate the subsystem using either digital or FM recording.

Figure C-3 summarizes the equipment used on each system. Note that both the digital and FM record/playback weights and complexity favor a real-time, FM, transmission system. The real-time system would be more compatible with the baseline system, since the anticipated procedure is to observe biological experiments which are initiated only during spacecraft passes over ground video receiving stations. The real-time system can use the present VHF uplink, but requires an S-Band transmitter and antenna for downlink. The S-Band transmitter replaces one of the baseline VHF transmitters. By using a real-time FM video system (Figure C-2), the A/D converter/conditioner, Digital Recorder Multiplexer and recorder are not required.

Although the input power requirements of the camera, control unit, and transmitter are high (97 watts), the short duty cycle (one minute per 96 minute orbit) has only minor impact on the baseline power system; i. e., 97 watt-minutes added to the 9964 watt-minutes per orbit of Mission I. The short peak demand is supplied by the battery.

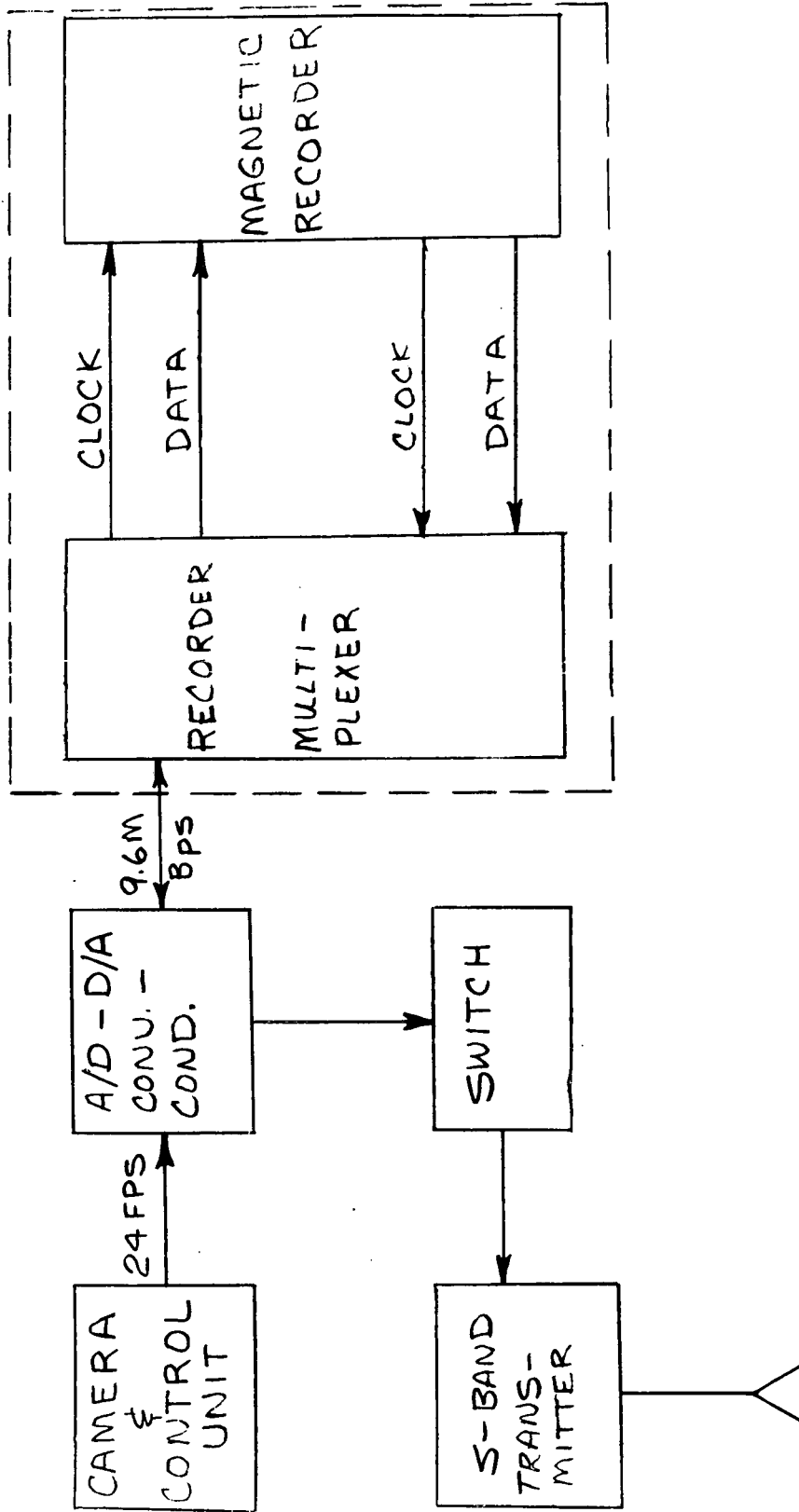


FIGURE C-1.- TV SYSTEM, DIGITAL

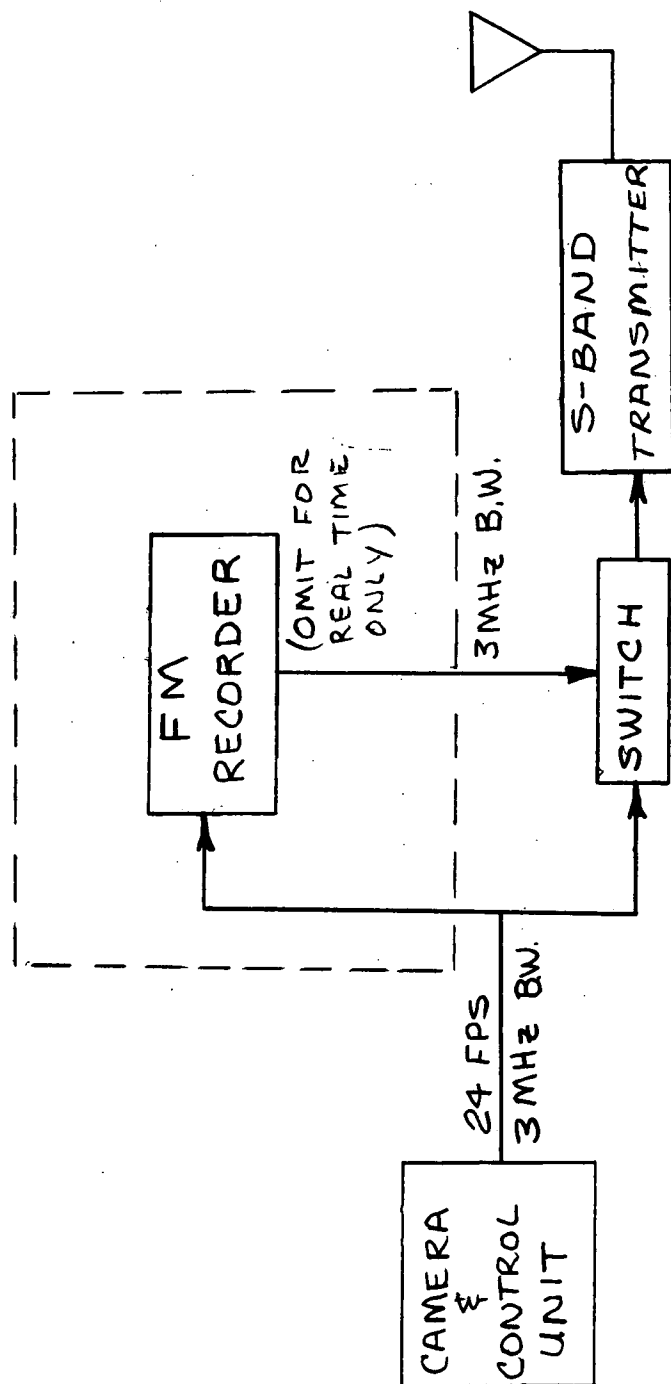


FIGURE C-2.- TV SYSTEM, FM

FIGURE C-3. - TV SYSTEM EQUIPMENT SUMMARY

Camera: GEC ED6038A and Control Unit			A/D Converter/Conditioner:		
Output - 24 frames/sec and 300 lines/frame video			Input - Analog video with line and frame sync. 24 frames/ sec.		
Horizontal & Vertical Sync. Pulses			Output - 9.6 MBps serial stream		
	<u>Camera</u>	<u>Control</u>			
Wt.	14 oz.	8.25 lbs	Wt.	5 lb. ₃	(est)
Volume	15 in. ³	1.72 in. ³	Vol.	10 in.	(est)
Power	15 watts		Power	2 watts	(est)
Digital Rec. Multiplexer:			Transmitter: SC1 MOD		
Wt.	.5 lb.	(est)	P/N 2208100-1		
Vol.	10 in. ³	(est)	Wt.	5 lbs.	
Power	2 watts	(est)	Vol.	100 in. ³	
			Power Input	72 watts	
			Power Output	10 watts	

Recorder:

Record Time: 1 minute
Storage Capacity: 546(10)⁶ Bits

	<u>Digital</u>	<u>FM</u>
Weight:	9 lbs (est)	38 lbs
Volume:	300 in. ³ (est)	1900 in. ³
Power:	9 watts (est)	85 watts

<u>Totals:</u>	<u>Digital</u>	<u>FM</u>	<u>Real-Time Only FM</u>
Weight:	22.25	51.25	12.25 lbs
Volume:	607	2187	287 in. ³
Power:	38	110	25 watts

3.0 S-BAND SYSTEM

The baseline Bioresearch Module has a data rate of 1.68 K bits per second and a T/M bandwidth of 30 KHz using a VHF system. Much higher data rates are achievable by changing systems. Since video transmission requires large bandwidths, it is feasible to define a wide bandwidth T/M system for the spacecraft. A 3 MHz bandwidth at a carrier frequency of 2300 MHz is chosen because of compatibility with video transmission.

Figure C-4 illustrates the S-Band subsystem. Existing baseline components such as the encoder, data storage, housekeeping, clock, decoder, signal conditioner, and power distribution systems are retained. The baseline T/M VHF transmitters, antennas and related hardware, and command receivers are replaced with S-Band components.

Table C-1 lists the three major components required by the spacecraft for S-Band. Table C-2 shows the technique used in determining the required power output of the S-Band T/M transmitter. Table C-3 summarizes the impact of the change from VHF to S-Band upon the baseline power system. Table C-4 summarizes the subsystem.

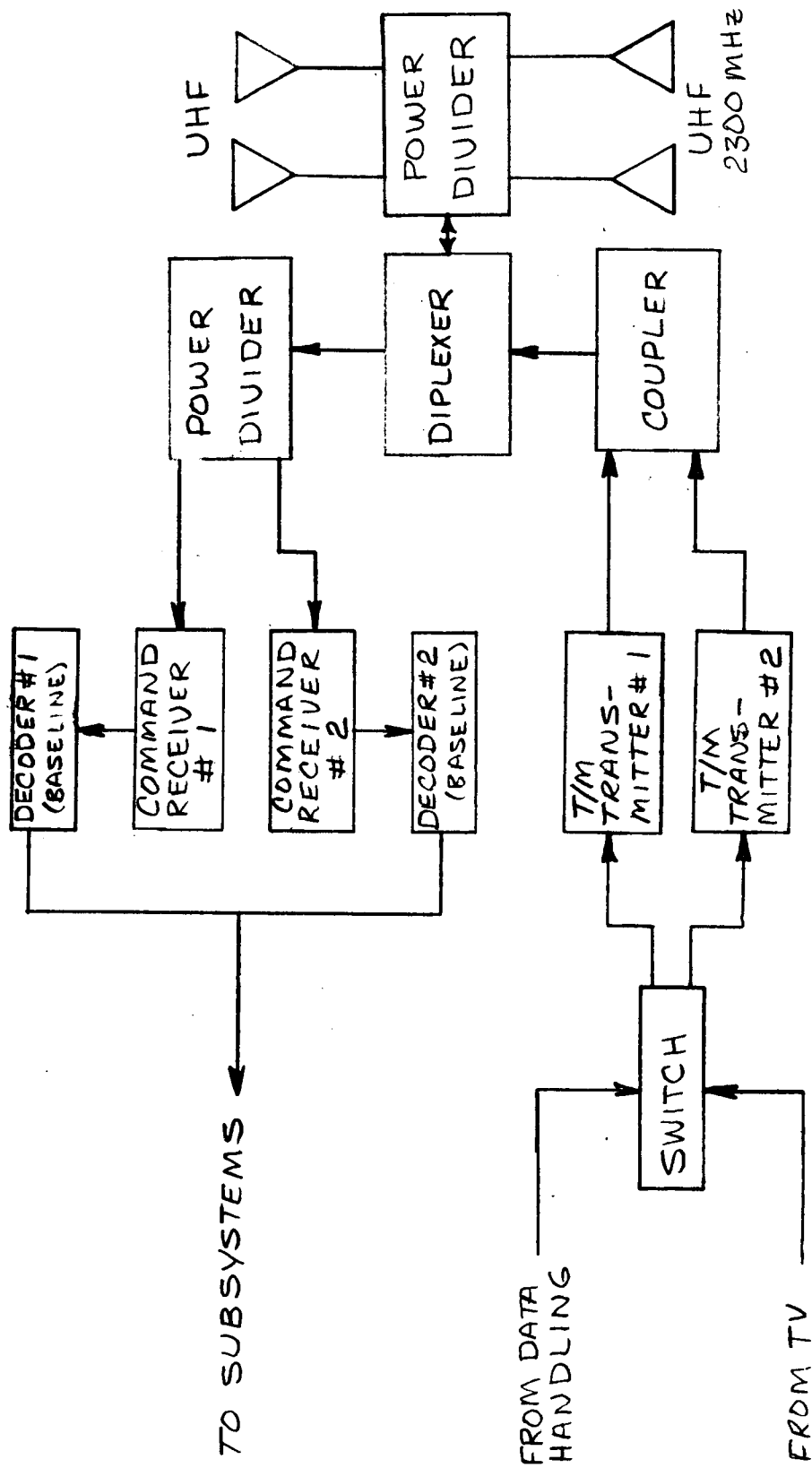


FIGURE C-4.— S BAND SYSTEM

TABLE C-1.- S-BAND COMMUNICATION EQUIPMENT

- Receiver: AVCO AED-303-2

Frequency: 2000-2400 MHz

Noise Figure: 7 Db

Power: 1.3 Watts, 28 VDC

Volume: 25 in.³ Weight: 1.5 Lbs.

- Transmitter: SCI MOD of P/N 2208100-1

Power Input: Est. 72 Watts minimum

Volume: 100 in.³ Weight: 4 Lbs.

S-BAND ANTENNA
(Designed to Fit Vehicle Envelope)

- Type: Four Cavity-backed Slots Spaced Around Spin Axis
- Pattern: Doughnut with Maxima 90° to Spin Axis

120° Beamwidth

+4 db gain at maxima

-6 db gain near vertical

TABLE C-2. - REQUIRED S-BAND TRANSMITTER OUTPUT POWER

Spacecraft:	Bioresearch Module Mode:	MOD TV 3 MHz BW at 24 frames/sec ^z
Mission:	I & II	Frequency: 2300 MHz ^z
		Range: 300 NM orbit 1200 NM Slant Range
<u>Parameter</u>	<u>Characteristic</u>	
1. Receiver Threshold		
10 log K		-198.6 dbm
10 log BW	B W = 3 MHz	64.8 db
10 log Te	TE = 2610 ^o K	<u>34.2 db</u>
Te = (NF - 1) 290 ^o K		Threshold = -99.6 dbm
= (10-1) 290 ^o K		
= 2610 ^o K		
2. Receiving Antenna	85 Ft. Dish	51.0 db
3. Polarization Loss		-3.0 db
4. Space Loss 37.8 db		-37.8 db
20 log NM	NM = 1200	61.6 db
20 log F	F = 2300 MHz ^z	<u>67.2 db</u>
	Space Loss =	-166.6 db
5. Vehicle Antenna	Omni Doughnut 120 ^o Beamwidth	+4.0 db
6. Vehicle System Loss		-5.0 db
7. Signal to Noise	Std. Coml. Quality	(-)20.0 db
8. Fade Margin		<u>0.0 db</u>
9. Transmitter Power	<u>10 Watts</u>	40.0 dbm

TABLE C-3.- IMPACT OF CHANGE FROM VHF TO S-BAND

	<u>Mission I</u>	<u>*Mission I(S)</u>	<u>Mission II</u>
<u>VHF System</u>			
Total S/C Pwr.Req'd., W	187	190	183
EOL Pwr. Avail., W	204	204	205
Power Margin, %	9	7	12
<u>S-Band System</u>			
Total S/C Pwr.Req'd., W	201	190	197
EOL Pwr. Avail., W	204	204	205
Power Margin, %	1.5	7	4
*I(S) Baseline Power System sized for VHF + S-Band (2% Duty Cycle)			

TABLE C-4.- S-BAND COMMUNICATION SUBSYSTEM

Missions I & II

- . Dual 2300 MHz Receiver - Transmitter System
- . Provides Command/TM/TV with failure redundancy
- . Provides 3MHz Bandwidth Analog TV Downlink
- . Simultaneous TM/TV using Both Transmitters
- . Antenna Pattern OMNI Doughnut with 120° Beamwidth
- . Transmitter Power Output 10 Watts

Power Input Approximately 72 Watts

REFERENCES

- C-1 "Modified Experiment Support Requirements to be
Evaluated under Task I-1", Enclosure (1) to NASA/Ames
letter PEF: 204-5(80L) dated 11 June 1971.

APPENDIX D

BIORESEARCH MODULE GROUND STATION SUPPORT EVALUATION

APPENDIX D
BIORESEARCH MODULE GROUND
STATION SUPPORT EVALUATION

1.0 INTRODUCTION

The STADAN and MSFN have recently been reorganized into one space-tracking and data acquisition network. This network is controlled by the Goddard Space Flight Center (GSFC). The top management officer for this network is Mr. Robert L. Owen, Chief of Network Engineering Division, phone number (301) 982-2816. For purposes of this evaluation, the STADAN and MSFN are considered separate networks. All NASA tracking facilities are listed in Table D-1 and are shown in Figure D-1. Figure D-2 shows the Space Tracking and Data Acquisition Network (STADAN) and Figure D-3 shows the Manned Space Flight Network (MSFN). Certain existing Eastern Test Range, Western Test Range and White Sands Missile Range support is included with the MSFN capabilities; however, no evaluation of these total range capabilities has been performed. Data presented in this study are based on References D-1 through D-4 and contact with NASA/GSFC personnel.

TABLE D-1. - NASA TRACKING FACILITIES* (STADAN & MSFN)

STATION	NETWORK	CODE NAME	TELEMETRY		COMMAND ANTENNA	TRACKING	
			SYSTEM	FREQUENCY			
Fairbanks, Alaska	STADAN	Alaska	85-FT Dish(2)	136 MHz 235 MHz 400 MHz 1700 MHz	SATAN (2)	MT, MOTS, GRARR	
NESC/CDA (Alaska)	STADAN	Alaska	40-FT Dish	136 MHz 400 MHz 1700 MHz	Disk-On-Rod		
Fort Myers, Fla.	STADAN	FTMYRS	SATAN Rec 9-yagi Rec	130-140 MHz	SATAN Dual-yagi Single-yagi	MT, MOTS	
Johannesburg, South Africa	STADAN	JOBURG	40-FT Dish SATAN Rec 8-yagi Rec	136 MHz 400 MHz 136 MHz 136 MHz	SATAN Dual-yagi	MT, MOTS	
Lima, Peru (Deactivated)	STADAN	LIMAPU					
Tananarive, Malagasy Republic (Madagascar)	STADAN/MSFN	MADGAR	40-FT Dish SATAN Rec Dual-yagi	136 MHz 400 MHz 136 MHz	SATAN	MT, MOTS, GRARR C-Band (MSFN)	
Barstov, Calif. (Caretaker Status)	STADAN	MOJAVE					

*Certain ETR, WTR, and DSIF stations support the Manned Space Flight Program as indicated.

TABLE D-1. - NASA TRACKING FACILITIES* (STADAN & MSFN) (Continued)

STATION	NETWORK	CODE NAME	TELEMETRY		COMMAND ANTENNA	TRACKING	
			SYSTEM	FREQUENCY			
Goddard Space Flight Center - NTTF	STADAN	NTTF	8-yagi SATAN	136 MHz 136 MHz	Dual-yagi		
Orroral, Australia	STADAN	ORORAL	85-FT Dish SATAN Rec 8-yagi	136 MHz 400 MHz 1700 MHz 136 MHz 136 MHz	Dual-yagi SATAN	MT, MOTS	
Quito, Ecuador	STADAN	QUITOE	40-FT Dish SATAN Rec 9-yagi (APC)	136 MHz 400 MHz 136 MHz 136 MHz	Dual-yagi, 123/148 MHz	MT, MTS	
Rosman, North Carolina	STADAN	ROSMAN	85-FT Dish(2) SATAN Rec 9-yagi Taco	136 MHz 400 MHz 1700 MHz 136 MHz	Disc-On-Rod SATAN	MOTS, GRARR	
Santiago, Calif.	STADAN	SNTAGO	40-FT X-Y Trk SATAN Rec 9-yagi Taco	136 MHz	Dual-yagi (123-148 MHz) SATAN	MT, MOTS, GRARR	
Cooby Creek, Australia (TOOWOOMBA)	STADAN	TOOMBA	16-yagi Taco Manual Control Range	148 MHz			
Winkfield, England	STADAN	WNKFLD	14-FT Dish SATAN Rec 8-yagi	400 MHz 136 MHz 136 MHz	Yagi Dual-yagi SATAN	MT, MOTS	

*Certain ETR, WTR, and DSIF stations support the Manned Space Flight Program as indicated.

TABLE D-1.1. - NASA TRACKING FACILITIES (STADAN & MSFN) (Continued)

STATION	NETWORK	CODE NAME	TELEMETRY		COMMAND ANTENNA	TRACKING	TV MONITOR
			SYSTEM	FREQUENCY			
Carnarvon, Australia (See MSFN Status)	STADAN (MSFN)	CARVON			Dual-yagi (STADAN)	GRARR (STADAN)	
Kauai, Hawaii (See MSFN Status)	STADAN (MSFN)	KAUAIH	9-yagi Rec, Twin yagi (STADAN)	136 MHz	Crossed-yagi 123/148 MHz (STADAN)		
USNS Vanguard	MSFN	VAN	30-FT Dish (USB) 12-FT Dish	225-2300 MHz 130-140 MHz 225-300 MHz	UHF/S-Band Circularly polarized steerable quad-helical array	C-Band	X
USNS Redstone	MSFN	RED	Same Equipment	as VAN			X
USNS Mercury	MSFN	MER	Same Equipment	as VAN			X
USNS Huntsville	MSFN	MER	12-FT Crossed Dipole	216-260 MHz	Same as Van		
Ascension Island	MSFN	ACN ASC	30-FT USB (Dual)	212-260 MHz 2270-2300 MHz	USB	USB, C-Band (ETR)	X
Bermuda	MSFN	BDA	30-FT USB (Dual uplink only)	212-260 MHz 2270-2300 MHz	USB UHF	USB, C-Band Azusa-Glotrac	X
Grand Canary Is.	MSFN	CYI	30-FT USB (Dual)	212-260 MHz 2270-2300 MHz	USB UHF	USB, C-Band	X

*Certain ETR, WTR, and DSIF stations support the Manned Space Flight Program as indicated.

TABLE D-1. - NASA TRACKING FACILITIES* (STADAN & MSFN) (Continued)

STATION	NETWORK	CODE NAME	TELEMETRY		COMMAND ANTENNA	TRACKING	TV MONITOR
			SYSTEM	FREQUENCY			
Grand Bahama Is. (Possibly Caretaker Status)	MSFN	GEM GBI	30-FT USB	2270-2300 MHz 212-260 MHz	USB UHF	USB, C-Band (ETR)	X
Goldstone, Calif.	MSFN/DSIF	GDS	85-FT USB (Dual)	2270-2300 MHz	USB	USB	X
Madrid, Spain	MSFN/DSIF	MAD	85-FT USB	2270-2300 MHz	USB	USB	X
Merritt Island, Florida	MSFN	MIL MLA	30-FT USB	212-260 MHz 2270-2300 MHz	USB UHF	USB, C-Band (ETR)	X
Guam, Marianas	MSFN	GWM	30-FT USB (Dual)	2270-2300 MHz	USB	USB	X
Guaymas, Mexico	MSFN	GYM	30-FT USB (Dual)	2270-2300 MHz	USB	USB	X
Kauai, Hawaii (See STADAN Status above)	MSFN (STADAN)	HAW	30-FT USB (Dual)	212-260 MHz 2270-2300 MHz	USB UHF	USB, C-Band	X
Corpus Christi, Texas	MSFN	TEX	30-FT USB	212-260 MHz 2270-2300 MHz	USB	USB	X
Honeysuckle Creek, Australia	MSFN/DSIF	HSK	85-FT USB	2270-2300 MHz	USB	USB	X
Antigua	MSFN/ETR	ANG ANT	30-FT USB	2270-2300 MHz 212-260 MHz	USB UHF (ETR)	C-Band (ETR)	X

*Certain ETR, WTR, and DSIF stations support the Manned Space Flight Program as indicated.

TABLE D-1. - NASA TRACKING FACILITIES* (STADAN & MSFN) (Concluded)

STATION	NETWORK	CODE NAME	TELEMETRY		COMMAND ANTENNA	TRACKING	TV MONITOR
			SYSTEM	FREQUENCY			
Cape Kennedy, Fla.	ETR (MSFN Support)	CNV			UHF	Offset Doppler, C-Band, Azusa-Glotrac	
Patrick Air Force Base, Florida	ETR (MSFN Support)	PAT			UHF	C-Band	
South Vandenberg, California	WTR (MSFN Support)	CAL	Telemetry Acq. Aid Antennas			C-Band	
Pretoria, South Africa	ETR (MSFN Support)	PRE				C-Band	
White Sands, N. Mexico	White Sands Missile Range (MSFN Support)	WHS	T/M Acq.	Aid Ant.		C-Band	
Goddard Space Flight Center	MSFN	NTTF	30-F USB	212-260 MHz 2270-2300 MHz	USB UHF	USB	X
Carnarvon, Australia (See STADAN Status above)	MSFN (STADAN)	CRO	30-FT USB (Dual)	212-260 MHz 2270-2300 MHz	USB UHF	USB, C-Band	X

* Certain ETR, WTR and DSIF stations support the Manned Space Flight Program as indicated.

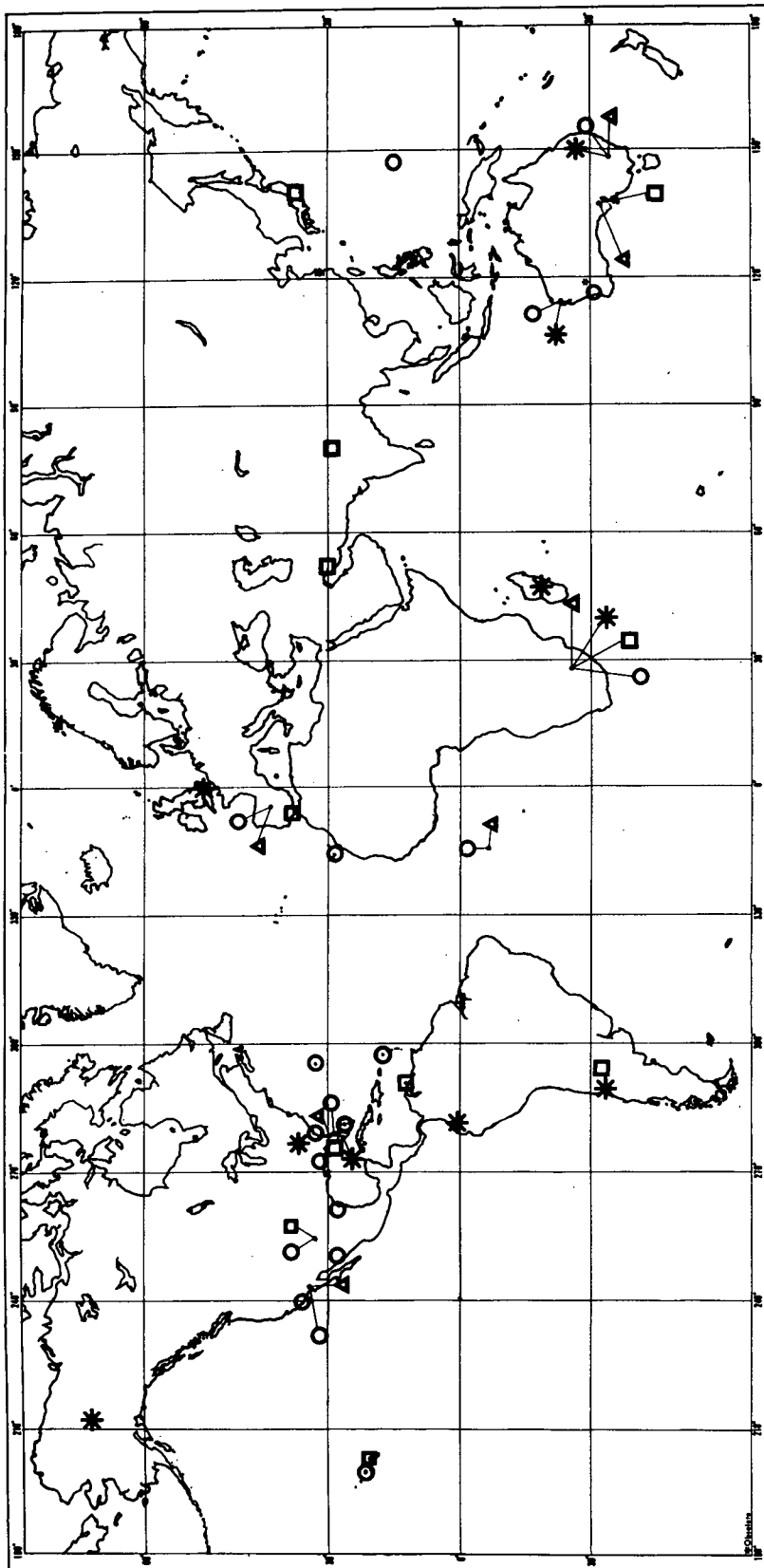
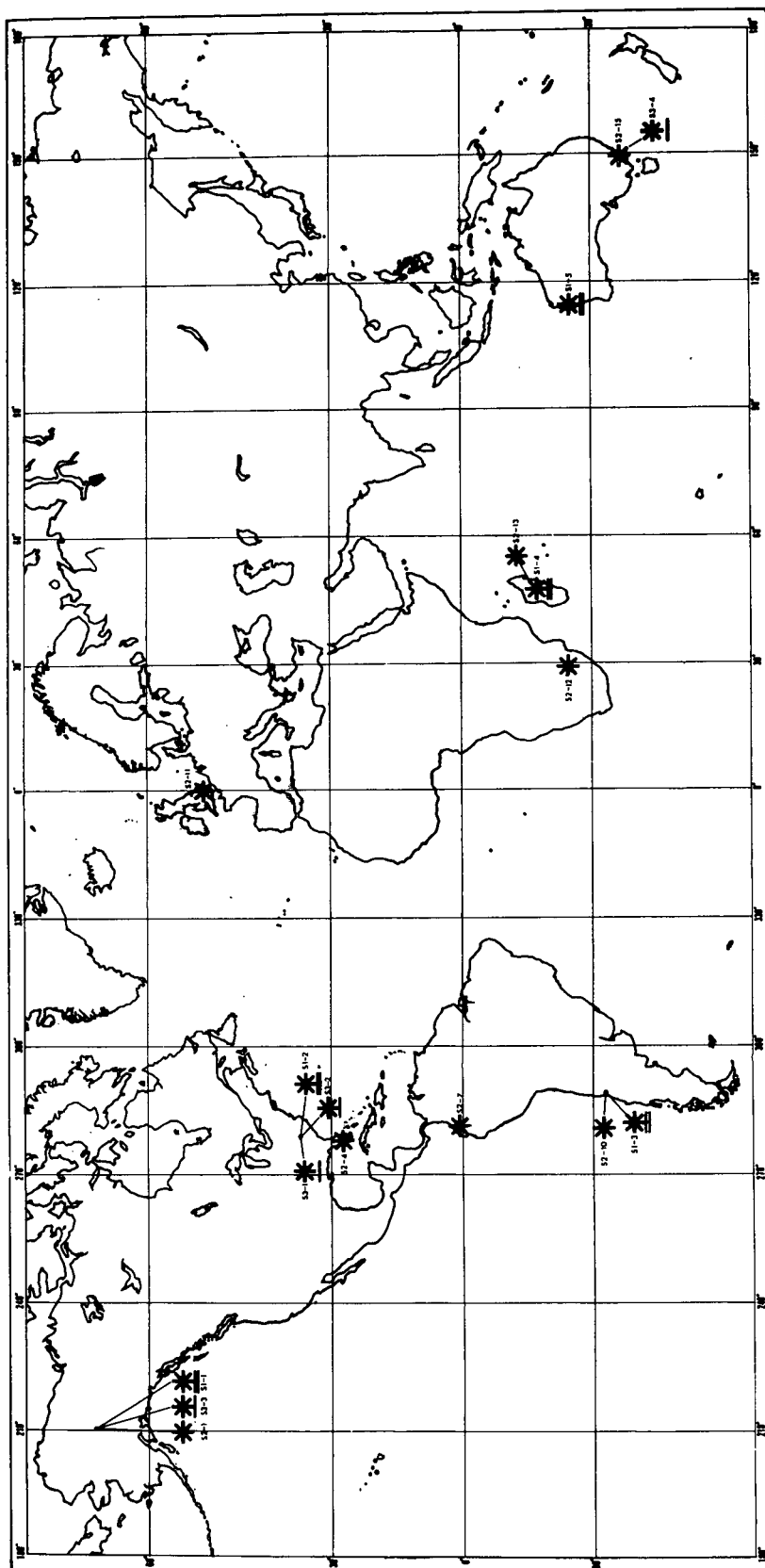


FIGURE D-1.- NASA TRACKING FACILITIES



*** RANGE AND RANGE RATE**
 S1 - 1 FAIRBANKS, ALASKA
 S1 - 2 ROSMAN, NORTH CAROLINA
 S1 - 3 SANTIAGO, CHILE
 S1 - 4 TANANARIVE, MALAGASY REE
 S1 - 5 CARNARVON, AUSTRALIA

*** MINITRACK**
S2 - 1 FAIRBANKS, ALASKA
S2 - 4 FORT MYERS, FLORIDA
S2 - 7 QUITO, ECUADOR
S2 - 10 SANTIAGO, CHILE

S2 - 11 WINKFIELD, ENGLAND
S2 - 12 JOHANNESBURG, R.S.A.
S2 - 13 TANANARIVE, MALAGASY REP.
S2 - 15 ORRORAL, AUSTRALIA

*** 85 FOOT ANTENNAS**
S3 - 1 ROSMAN, NORTH CAROLINA
S3 - 2 ROSMAN, NORTH CAROLINA
S3 - 3 FAIRBANKS, ALASKA (2)
S3 - 4 ORRORAL, AUSTRALIA

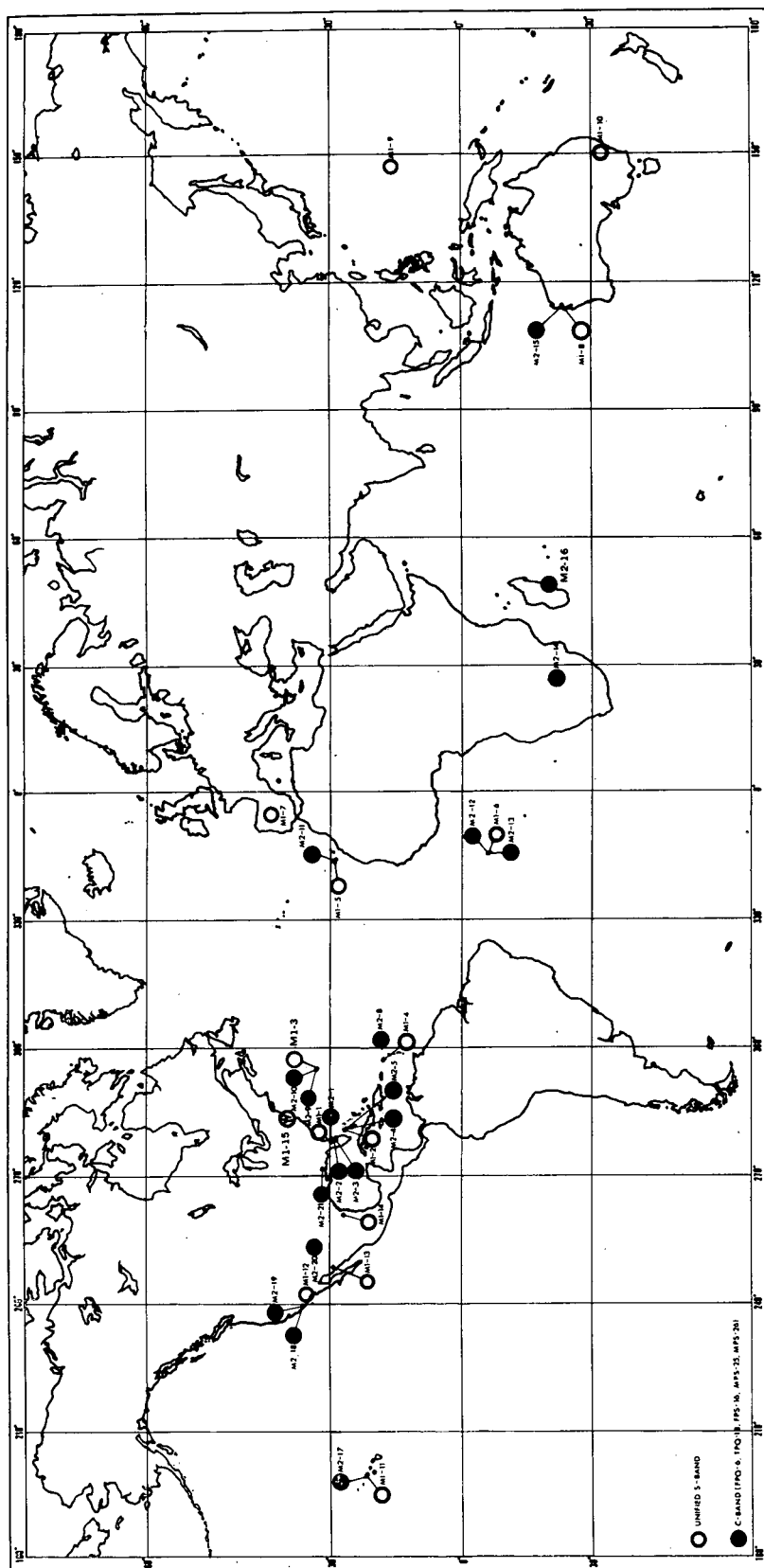


FIGURE D-3.- MANNED SPACE FLIGHT NETWORK

UNIFIED S-BAND NETWORK

- M1 - 1 MERRITT ISLAND, FLORIDA
- M1 - 2 GRAND BAHAMA
- M1 - 3 BERMUDA
- M1 - 4 ANTIGUA
- M1 - 5 CANARY ISLANDS
- M1 - 6 ASCENSION ISLAND
- M1 - 7 MADRID, SPAIN
- M1 - 8 CARNARVON, AUSTRALIA
- USNS VANGUARD } NOT SHOWN
- USNS REDSTONE }

C-BAND RADARS

- M2 - 9 GUAM
- M2 - 10 HONEYSUCKLE CREEK, AUSTRALIA
- M2 - 11 KAUAI, HAWAII
- M2 - 12 GOLDSTONE, CALIFORNIA
- M2 - 13 GUAYMAS, MEXICO
- M2 - 14 CORPUS CHRISTI, TEXAS
- M2 - 15 GODDARD SPACE FLIGHT CENTER (NTFI)
- USNS MERCURY } NOT SHOWN
- USNS HUNTSVILLE }

C-BAND RADARS

- M2 - 1 MERRITT ISLAND, FLORIDA
- M2 - 2 PATRICK AFB, FLORIDA
- M2 - 3 CAPE KENNEDY, FLORIDA
- M2 - 4 GRAND BAHAMA
- M2 - 5 GRAND BAHAMA
- M2 - 6 ANTIGUA
- M2 - 7 BERMUDA
- M2 - 8 BERMUDA
- M2 - 9 BERMUDA
- M2 - 10 BERMUDA
- M2 - 11 CANARY ISLANDS
- USNS VANGUARD } NOT SHOWN
- USNS REDSTONE }

C-BAND RADARS

- M2 - 12 ASCENSION ISLAND
- M2 - 13 ASCENSION ISLAND
- M2 - 14 PRETORIA, SOUTH AFRICA
- M2 - 15 CARNARVON, AUSTRALIA
- M2 - 16 TANANARIVE, MALAGASY
- M2 - 17 KAUAI, HAWAII
- M2 - 18 VANDENBERG A.F.B., CALIFORNIA
- M2 - 19 POINT ARGUELLO, CALIFORNIA
- M2 - 20 WHITE SANDS, NEW MEXICO
- USNS MERCURY } NOT SHOWN
- USNS HUNTSVILLE }

2.0 GROUND STATION SUPPORT OF SPACECRAFT TELEVISION

2.1 ANALYSIS

The ground station capabilities to support real time or recorded television were investigated and evaluated. The 137 MHz telemetry carrier frequency used in the baseline Bioresearch Module, which is that frequency assigned to the U.S. for space research telemetering by agreements reached by the Extraordinary Administrative Radio Conference of 1963, is allocated a nominal bandwidth of 30 KHz with a maximum bandwidth of 90 KHz. The requirement for real time or recorded television will require 1-3 MHz bandwidth depending on 100 to 300 line resolution, respectively.

Recent inquiries at GSFC (Frequency Control Officer) indicate the wideband requirements for television on the 137 MHz carrier frequency would not be acceptable for two reasons:

- (1) Currently none of the STADAN ground stations have the capability to handle wideband transmission. This capability could be obtained by purchasing additional equipment.
- (2) Other projects, such as the meteorological satellites operating in the 137-138 MHz frequency range would probably object to the transmission interference during Bioresearch Module wideband transmission.

The Administration Radio Conference concluded a world conference on August 11, 1971. New significance was added to the 136-138 MHz carrier frequency range in that the United States requested and obtained more bandwidth. The 138 MHz to 143.6 MHz carrier frequency was allocated to space research in region 2, in which North and South America are located. Therefore, a possibility exists for obtaining wideband (television) data utilizing the current telemetry system proposed for the Bioresearch Module providing that the STADAN wideband capabilities are obtained. A request to GSFC at this time to use wideband transmission in the 138 to 143.6 MHz range would probably be a first.

The Unified S-Band (USB) system, Figure D-3, provides tracking, ranging, command, telemetry and voice communications for the Apollo missions. The spacecraft transponder transmitter can be frequency-modulated for the transmission of real time television information or recorded data. GSFC has indicated that the USB system would be suitable for the Bioresearch Module wideband (television) requirement provided the carrier frequency is in the S-Band region, 2270-2300 MHz.

2.2 CONCLUSION

STADAN ground station support capabilities for real time or recorded television do not currently exist due to the wideband (1 to 3 MHz) requirement. A possibility of STADAN wideband ground station support may occur in the future with the allocation of the 138 to 143.6 MHz carrier frequency to space research and provided STADAN acquires wideband ground station equipment.

3.0 DESIGN IMPACT OF USING STADAN AND/OR MSFN

3.1 ANALYSIS

The design impact of using the Manned Space Flight Net either in addition to or in lieu of STADAN for ground station support was analyzed. It was assumed that the MSFN shall be available to support missions requiring television monitoring of the experiment.

Three configurations were considered which use the STADAN and/or MSFN ground station support for the Bioresearch Module telemetry, tracking, and command systems with wideband capability (television):

- (1) Telemetry carrier frequency 137 MHz, tracking carrier frequency 136 MHz, command carrier frequency 148 MHz, and wideband (television) capability using S-Band, 2270-2300 MHz.
- (2) Telemetry carrier frequency of 137 MHz with wideband ground station capability, tracking carrier frequency of 136 MHz and command carrier frequency of 148 MHz.
- (3) Telemetry, tracking and command carrier frequencies, including television monitoring, using S-Band.

The prime acquisition sites for telemetry, tracking and command for configuration (1) are as follows:

<u>Telemetry, Tracking and Command</u>	<u>Wideband Telemetry S-Band</u>
Roseman, North Carolina (STADAN)	Merritt Island, Florida (MSFN)
Quito, Ecuador (STADAN)	---
Santiago, Chili (STADAN)	---
Johannesburg, South Africa (STADAN)	---
Tananarive, Malagasy Rep. (STADAN)	---
Carnarvon, Australia (STADAN)	Carnarvon, Australia (MSFN)

The prime acquisition sites for telemetry, tracking and command for configuration (2) are as follows:

Tananarive, Malagasy Republic (STADAN)
Quito, Ecuador (STADAN)

Rosman, North Carolina (STADAN)

Santiago, Chili (STADAN)

Orroral, Australia (STADAN)

The prime acquisition sites for telemetry, tracking and command including television monitoring for configuration (3) are as follows:

Ascension Island (MSFN)

Bermuda (MSFN)

Grand Canary, Island (MSFN)

Goldstone, California (MSFN)

Merritt Island, Florida (MSFN)

Guam, Marianas (MSFN)

Guaymas, Mexico (MSFN)

Kauai, Hawaii (MSFN)

Corpus Christi, Texas (MSFN)

Honeysuckle Creek, Australia (MSFN)

Antigua (MSFN)

3.2 CONCLUSION

Using the Manned Space Flight Net in addition to or in lieu of STADAN for ground station support will require that the Bioresearch Module contain provisions for S-Band capability (2270 to 2300 MHz carrier frequency). The Bioresearch Module using the 136-137 MHz carrier frequency for tracking and telemetry, and 148 MHz carrier frequency for command functions cannot be supported by the MSFN. The Manned Space Flight Network facilities require that the spacecraft transponder operate in the 2270-2300 MHz frequency range.

REFERENCES

- D-1. "MSFN Equipment Allocations Handbook", Part I, Systems Sequence, NASA MG-403 revised 31 January 1971.
- D-2. "MSFN Equipment Allocations Handbook", Part II, Site Sequence, NASA MG-403 revised 31 January 1971.
- D-3. "Telemetry Data Processing Facility Project Support Handbook", Volume I, NASA/GFSC Report, July 1970.
- D-4. "Telemetry Data Processing Facility Project Support Handbook", Volume II, NASA/GFSC Report, July 1970.

APPENDIX E

VARIABLE SPIN CONTROL ANALYSIS

1.0 SPACECRAFT STABILITY STUDY

A dynamic stability study is presented for the spacecraft with its attached, flexible booms.

1.1 IDEALIZED SYSTEM

The idealized system to be analyzed is shown in Figure E-1.

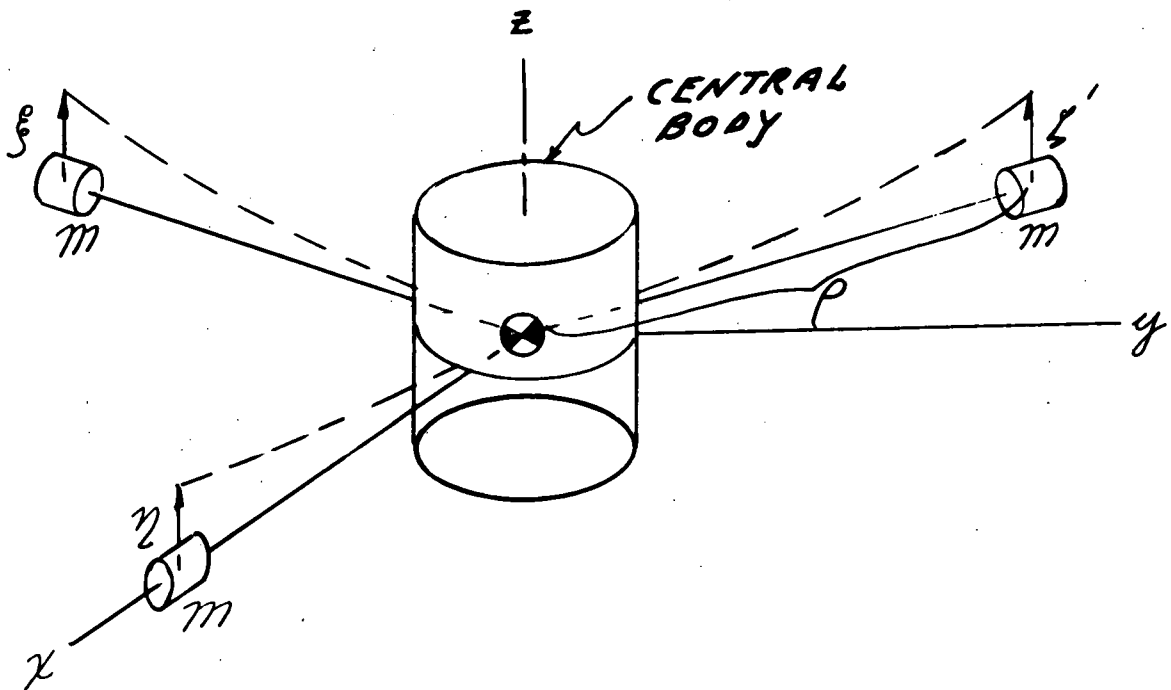


FIGURE E-1. - IDEALIZED SPACECRAFT

It is to be noted that

- a) x , y , z are right handed, non-inertial axes. They are fixed in the body, along principal axes of the "rigid body".

- b) I_x, I_y, I_z are principal moments of inertia about the principal axes x, y, z , respectively.
- c) $I_{xy} = I_{xz} = I_{yz} = 0$ are products of inertia
- d) m = equivalent tip mass
- e) $\zeta = -\eta, \xi = -\eta$
- f) $\bar{i}, \bar{j}, \bar{k}$ are unit vectors, along x, y, z respectively and are time-dependent.
- g) The center of mass of the rigid body is in an inertial frame.

1.2 EQUATIONS OF MOTION

The equations of motion will now be written. They consist of the motion of the booms, relative to the central hub, and the "rigid body motion" of the central body with its attached booms.

1.2.1 Kinetic Energy. - The kinetic energy is given by

$$T = \frac{1}{2} [I_x \omega_x^2 + I_y \omega_y^2 + I_z \omega_z^2 + \frac{1}{2} m [\bar{V}_\eta' \cdot \bar{V}_\eta' + \bar{V}_\zeta' \cdot \bar{V}_\zeta' + \bar{V}_\xi' \cdot \bar{V}_\xi'] \quad (1.1)$$

in which, $\bar{V}_\eta' = \dot{\eta} \bar{k} + \bar{\omega} \times \bar{r}_1$

$$\bar{V}_\zeta' = \dot{\zeta} \bar{k} + \bar{\omega} \times \bar{r}_2$$

$$\bar{V}_\xi' = \dot{\xi} \bar{k} + \bar{\omega} \times \bar{r}_3 \quad (1.2)$$

and further, $\bar{r}_1 = \rho \bar{i}$

$$\bar{r}_2 = \rho [-\sin \theta \bar{i} + \cos \theta \bar{j}] - \eta \bar{k}$$

$$\bar{r}_3 = \rho [-\sin \theta \bar{i} - \cos \theta \bar{j}] - \eta \bar{k} \quad (1.3)$$

1.2.2 Potential Energy. - The spacecraft is in a state of "gravity free" motion as far as its center of mass is concerned. Thus, we are concerned only with the potential energy of the elastic booms.

Hence,

$$V = \frac{1}{2} k_n \eta^2(z) \quad (1.4)$$

in which k_n is an effective spring rate, given by

$$k_n = \frac{3EI_\eta}{l^3} \left[1 + \frac{m\rho \omega_z^2 l^2}{2EI_\eta} \right] \quad (1.5)$$

1.2.3 Boom Equations. - The equations of motion of the booms are given through the application of Lagrange's equations in conjunction with expressions (1.1) through 1.5).

Lagrange's equations are given by

$$\frac{d}{dt} \left[\frac{\partial T}{\partial \dot{\eta}} \right] - \frac{\partial T}{\partial \eta} + \frac{\partial V}{\partial \eta} + \frac{\partial f}{\partial \eta} = 0 \quad (1.6)$$

If the indicated operations are carried out, there is obtained

$$\ddot{\eta} + \frac{c}{m} \dot{\eta} + \omega_\eta^2 \eta = \frac{\rho}{3} [\dot{\omega}_y (1 + 2 \sin \theta) - \omega_x \omega_z (1 + 2 \sin \theta)] \quad (1.7)$$

1.2.4 "Rigid Body" Equations. - The "Rigid Body" equations are given by

$$\dot{\bar{H}} = 0 \quad (1.8)$$

in which the derivative is relative to an inertial frame.

Now \bar{H} is divided into two portions as follows:

$$\bar{H} = \bar{H}_R + \bar{H}_E \quad (1.9)$$

So that,

$$\dot{\bar{H}}_R = -\dot{\bar{H}}_E = -\left\{ \sum_{i=1}^3 \bar{r}_i \times m \ddot{\bar{r}}_i \right\} \quad (1.10)$$

Further,

$$\bar{H}_R = I_x \omega_x \bar{I} + I_y \omega_y \bar{J} + I_z \omega_z \bar{K} \quad (1.11)$$

By equations (1.8) through (1.11) and remembering that

$$\dot{\bar{H}} = \frac{d}{dt} (\bar{H}_{xyz}) + \bar{\omega} \times \bar{H} \quad (1.12)$$

we arrive at the following Scalar equations of motion:

$$\dot{\omega}_x + \omega_z \omega_y \left(\frac{I_z - I_y}{I'_x} \right)' = 0 \quad (1.13)$$

$$\dot{\omega}_y + \omega_x \omega_z \left(\frac{I_x - I_z}{I'_y} \right)' = \frac{m\rho}{I'_y} \left[\ddot{\eta} (1 + 2 \sin \theta + \eta \omega_z^2 (1 + 2 \cos \theta)) \right] \quad (1.14)$$

$$\dot{\omega}_z + \omega_y \omega_x \left(\frac{I_y - I_x}{I'_z} \right)' = 0 \quad (1.15)$$

in which,

$$\left. \begin{aligned} I'_x &= I_x + 2 m \rho^2 \cos^2 \theta \\ I'_y &= I_y + m \rho^2 (1 + 2 \sin^2 \theta) \\ I'_z &= I_z + 3 m \rho^2 \\ (I_z - I_y)' &= I_z - I_y + 2 m \rho^2 \cos^2 \theta \\ (I_x - I_z)' &= I_x - I_z - m \rho^2 (1 + 2 \sin^2 \theta) \\ (I_y - I_x)' &= I_y - I_x + 3 m \rho^2 \end{aligned} \right\} \quad (1.16)$$

1.3 STABILITY CRITERION

We take ω_z to be constant and consider the stability associated with equations (1.7), (1.13) and (1.14).

We take solutions of the equations (1.7), (1.13) and (1.14) to be of the form

$$\left. \begin{aligned} \omega_y &= y e^{\lambda t} \\ \omega_x &= x e^{\lambda t} \\ \eta &= N e^{\lambda t} \end{aligned} \right\} \quad (1.17)$$

and the determinant of the set of the resulting algebraic equations is:

$$\begin{vmatrix} \lambda & \frac{\omega_z (I_z - I_y)'}{I_x'} & 0 \\ \frac{\omega_z (I_x - I_z)'}{I_y'} & \lambda & \frac{-m\rho}{I_y'} [\lambda^2 (1 + 2 \sin \theta) + \omega_z^2 (1 + 2 \cos \theta)] \\ \frac{\omega_z}{3} (1 + 2 \sin \theta) & \frac{-\rho}{3} \lambda (1 + 2 \sin \theta) & (\lambda^2 + \frac{C\lambda}{m} + \omega_n^2) \end{vmatrix} = 0 \quad (1.18)$$

The frequency equation results from the expansion of this determinant. It is of the 4th degree in λ , and may be written as

$$\sum_{i=0}^{i=n=4} a_i \lambda^{n-i} = 0 \quad (1.19)$$

The stability criteria are:

$$a_i > 0 \quad (i = 0, 1, 2, 3, 4) \quad (1.20)$$

Together with

$$a_3(a_1 a_2 - a_0 a_3) - a_1^2 a_4 > 0 \quad (1.21)$$

These inequalities must be satisfied for dynamic stability.

1.4 STABILITY PROFILES

For this spacecraft, $I_x = I_y$ and further we define

$$\left. \begin{aligned} \frac{I_z}{I_x} - 1 &= \alpha \\ \frac{m\rho^2}{I_x} &= \beta \\ \frac{\omega_z}{\omega_n} &= Z \end{aligned} \right\} \quad (1.22)$$

With these definitions and the stability criteria noted by (1.20) and (1.21), we may show various stability profiles. These appear in Figure E-2 for four values of β .

1.7 DEFINITION OF TERMS

T = Kinetic energy

E = Young's Modulus

I_η = 2nd moment of the boom cross-section

ρ = distance from the spacecraft center of mass to the mass, m

l = structural length of a boom.

$\omega_x, \omega_y, \omega_z$ = angular velocity components of the spacecraft, along x, y, z, respectively.

f = dissipation function for a boom.

θ = angle between the booms and specific axes (Figure E-1)

\bar{H}_E, \bar{H}_R = angular momenta of the elastic and rigid portions of the spacecraft, respectively.

V = potential energy

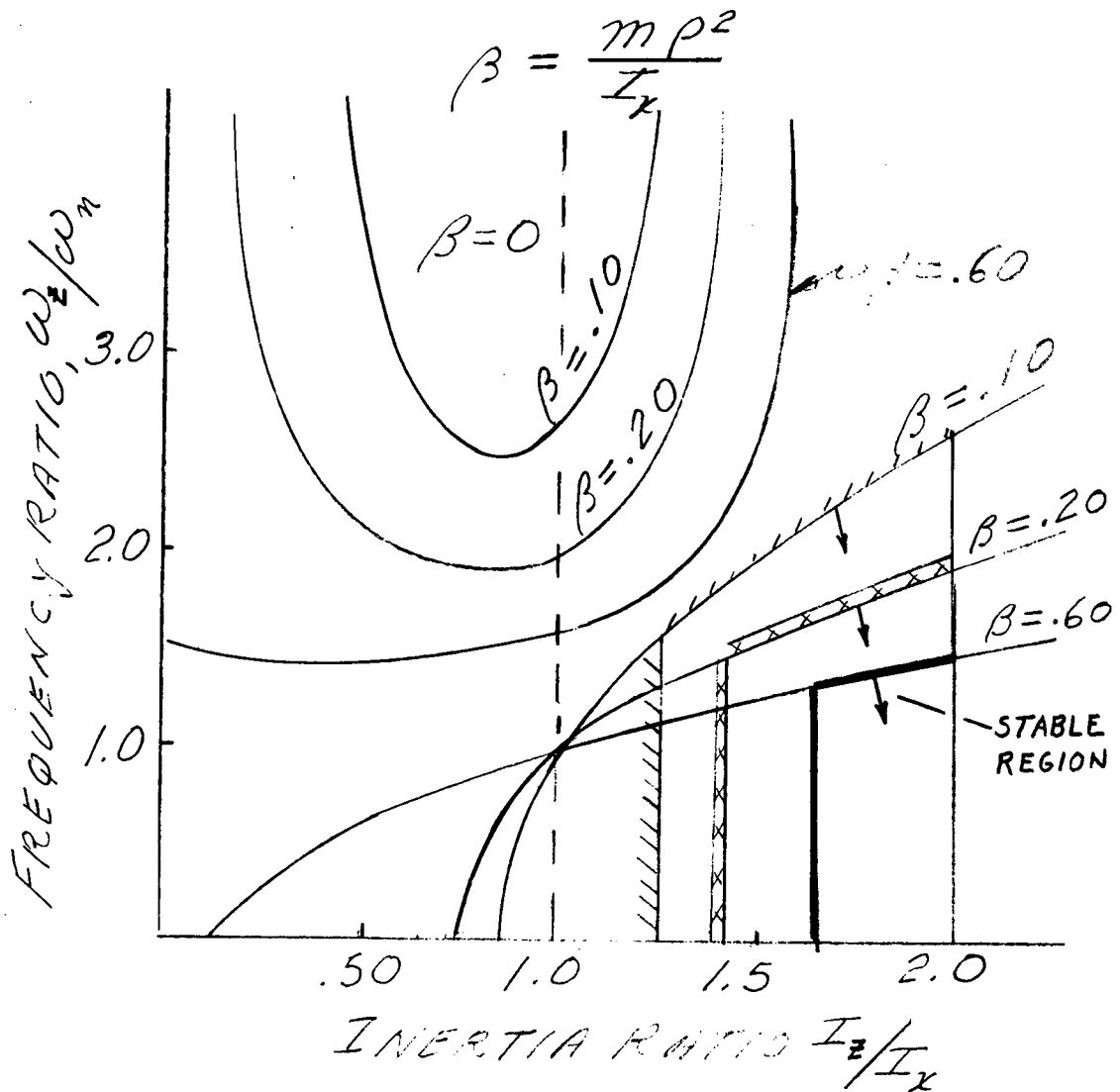


FIGURE E-2. - STABILITY PROFILES

2.1.1 Angular Velocity and Acceleration. - The angular velocity vector is given by

$$\begin{aligned}\bar{\omega} = & (\dot{\theta} \cos \psi + \dot{\phi} \sin \theta \sin \psi) \bar{i} \\ & + (-\dot{\theta} \sin \psi + \dot{\phi} \sin \theta \cos \psi) \bar{j} \\ & + (\dot{\phi} \cos \theta + \dot{\psi}) \bar{k}\end{aligned}\quad (2.1)$$

We will not allow the cone-angle velocity ($\dot{\theta}$) to grow large, so that we express $\bar{\omega}$ by

$$\begin{aligned}\bar{\omega} = & (\dot{\phi} \sin \theta \sin \psi) \bar{i} \\ & + (\dot{\phi} \sin \theta \cos \psi) \bar{j} \\ & + (\dot{\phi} \cos \theta + \dot{\psi}) \bar{k}\end{aligned}\quad (2.2)$$

The angular acceleration is approximated by

$$\dot{\bar{\omega}} = \dot{\phi} \dot{\psi} \sin \theta (\cos \psi \bar{i} - \sin \psi \bar{j}) \quad (2.3)$$

in which products in $\dot{\phi} \dot{\theta}$, and the terms $\ddot{\phi}$, $\ddot{\psi}$ have been ignored in comparison to the terms in expression (2.3).

2.1.2 Linear Acceleration. - Let the particle, P, be located by the position vector \bar{r} . Then the linear velocity of the point P is given by

$$\dot{\bar{r}} = \dot{\bar{r}}_{x'y'z'} + \bar{\omega} \times \bar{r} \quad (2.4)$$

The acceleration is given by

$$\ddot{\bar{r}} = \left[\ddot{\bar{r}}_{x'y'z'} + 2 \bar{\omega} \times \dot{\bar{r}}_{x'y'z'} + \bar{\omega} \times \bar{r} + \bar{\omega} \times (\bar{\omega} \times \bar{r}) \right] \quad (2.5)$$

in which x', y', z' is the non-inertial body axes.

Now

$$\ddot{\bar{r}}_{x'y'z'} = \ddot{\bar{r}}_{x_1'y_1'z_1'} = 0 \quad (2.6)$$

because the particle is stationary relative to the body. Thus,

$$\ddot{\vec{r}} = \dot{\vec{\omega}} \times \vec{r} + \vec{\omega} \times (\vec{\omega} \times \vec{r}) \quad (2.7)$$

This is the linear acceleration of the particle.

If we expand Equation (2.7) we obtain

$$\ddot{\vec{r}} = [\dot{\vec{\omega}} \times \vec{r} + \vec{\omega} (\vec{\omega} \cdot \vec{r}) - \vec{r} (\vec{\omega} \cdot \vec{\omega})] \quad (2.8)$$

We now orient the particle so that

$$r_{y'} \equiv 0$$

and substitute Equations (2.2) and (2.3) into (2.8). We then obtain

$$\ddot{\vec{r}} = \ddot{\vec{r}}(\phi, \psi, \dot{\phi}, \dot{\psi}, r_{x'}, r_{z'}) \quad (2.9)$$

From the constance of angular momentum, it is easy to show that

$$\left. \begin{aligned} \omega_{0z} \cos \theta &= \dot{\phi} \cos \theta + \dot{\psi} \\ \omega_{0z} \frac{I_z}{I_{x'}} \sin \theta \sin \psi &= \dot{\phi} \sin \theta \sin \psi \\ \omega_{0z} \frac{I_z}{I_{y'}} \sin \theta \cos \psi &= \dot{\phi} \sin \theta \cos \psi \end{aligned} \right\} \quad (2.10)$$

From which we find that

$$\left. \begin{aligned} \dot{\phi} &= \omega_{0z} \frac{I_z}{I_x} \\ \dot{\psi} &= \omega_{0z} \cos \theta [1 - I_z/I_x] \end{aligned} \right\} \quad (2.11)$$

in which we have used the fact that

$$I_{x'} = I_{y'} = I_x$$

We note also that ω_{0z} refers to the original spin speed of the body about the z-axis.

We can now express Equation (2.2) as

$$\begin{aligned}\bar{\omega} = \omega_{0z} \frac{I_z}{I_x} (\sin \theta \sin \psi \bar{i} \\ + \sin \theta \cos \psi \bar{j}) + \omega_{0z} \cos \theta \bar{k}\end{aligned}\quad (2.12)$$

and Equation (2.3) becomes

$$\dot{\bar{\omega}} = \omega_{0z}^2 \frac{I_z}{I_x} \cos \theta \sin \theta (1 - I_z/I_x) \quad (2.13)$$

If we now take θ to be small and take the spin-rate to be constant, i.e.,

$$\left. \begin{aligned}\sin \theta &\approx \theta \\ \cos \theta &\approx 1 \\ \psi &= \dot{\psi} t\end{aligned}\right\} \quad (2.14)$$

We find the linear acceleration of point P to be

$$\begin{aligned}\ddot{\bar{r}} = \omega_{0z}^2 \left\{ \left[r_z \left(\frac{I_z}{I_x} \right)^2 \theta \sin \dot{\psi} t + r_{x'} \right] \bar{i} \right. \\ \left. + \left[r_z' \left(\frac{I_z}{I_x} \right)^2 \theta \cos \dot{\psi} t \right] \bar{j} \right. \\ \left. + \left[r_{x'} \left(\frac{I_z}{I_x} \right) \theta \sin \dot{\psi} t (2 - I_z/I_x) \right] \bar{k} \right\}\end{aligned}\quad (2.15)$$

The terms above may be combined in scalar form to obtain steady-state terms (constant in time) and a time varying term. This latter term is the one of interest. It is

$$\ddot{\bar{r}}_t = \omega_{0z}^2 \left(\frac{I_z}{I_x} \right) \theta r_{x'} (2 - I_z/I_x) \sin \dot{\psi} t$$

Its maximum value is

$$\left| \ddot{\bar{r}}_t \right|_{\text{MAX}} = \omega_{0z}^2 \left(\frac{I_z}{I_x} \right) \theta r_{x'} (2 - I_z/I_x) \quad (2.16)$$

2.1.3 Maximum Allowable $\theta = \theta_1$. - The allowable cone angle θ_1 , is governed by the maximum allowable linear acceleration. This maximum allowable value is set at

$$|\ddot{r}_t|_{MAX} = 3 \times 10^{-3} g \quad (2.17)$$

From this value and Equation (2.16) we can solve for the maximum allowable angle,

$$\theta_1 = \frac{|\ddot{r}_t|}{\omega_{0z}^2 \left(\frac{I_z}{I_x} \right) r_x' (2 - I_z/I_x)} \quad (2.18)$$

Values of this angle are tabulated for various spin speeds as follows:

TABLE E-1. - MAXIMUM ALLOWABLE CONE ANGLE

SPIN SPEED ω_{0z}	CONE HALF-ANGLE θ_1
1.5 Radians/sec.	2.46°
5.83 Radians/sec.	.16°

Other data are

$$I_z = 7.81 \text{ slug-ft}^2$$

$$I_x = 18.75 \text{ slug-ft}^2$$

$$r_x' = 1.5 \text{ ft.}$$

2.2 TIME HISTORY

We now predict the time required by the spacecraft to develop the cone half-angle, θ_1 . We make this prediction on the basis of the dissipation of energy through the elastic motion resulting from the coning motion. Both the main body and the booms will contribute to this energy dissipation.

2.2.1 Energy Dissipation Relationship. - The drift angle rate, and the time rate of change of the kinetic energy of the spacecraft are related as follows:

$$\dot{T} = \dot{\theta} \frac{H^2}{I_z} (I_z/I_x - 1) \sin \theta \cos \theta \quad (2.19)$$

The energy dissipated in the elastic body due to hysteresis damping is, for each cycle of stress

$$\iiint_V \frac{\gamma \sigma^2}{2 E t_0} dV \quad (2.20)$$

In which γ is the hysteresis damping factor, σ is the repeated bending stress, E is Young's modulus and t_0 is the period of the repeated stress cycle, and V is the volume of the material which is undergoing repeated stress.

The energy dissipation (2.20) causes a change in T as given by Equation (2.19). Thus, for each cycle,

$$\dot{\theta} \frac{H^2}{I_z} \left(\frac{I_z}{I_x} - 1 \right) \sin \theta \cos \theta = - \iiint_V \frac{\gamma \sigma^2}{2 E t_0} dV \quad (2.21)$$

The problem now becomes one of determining σ and t_0 . The previous analyses of Section 2.1 furnishes information for obtaining both of these items.

2.2.2 Boom Bending. - If we now consider the system shown below, we may write

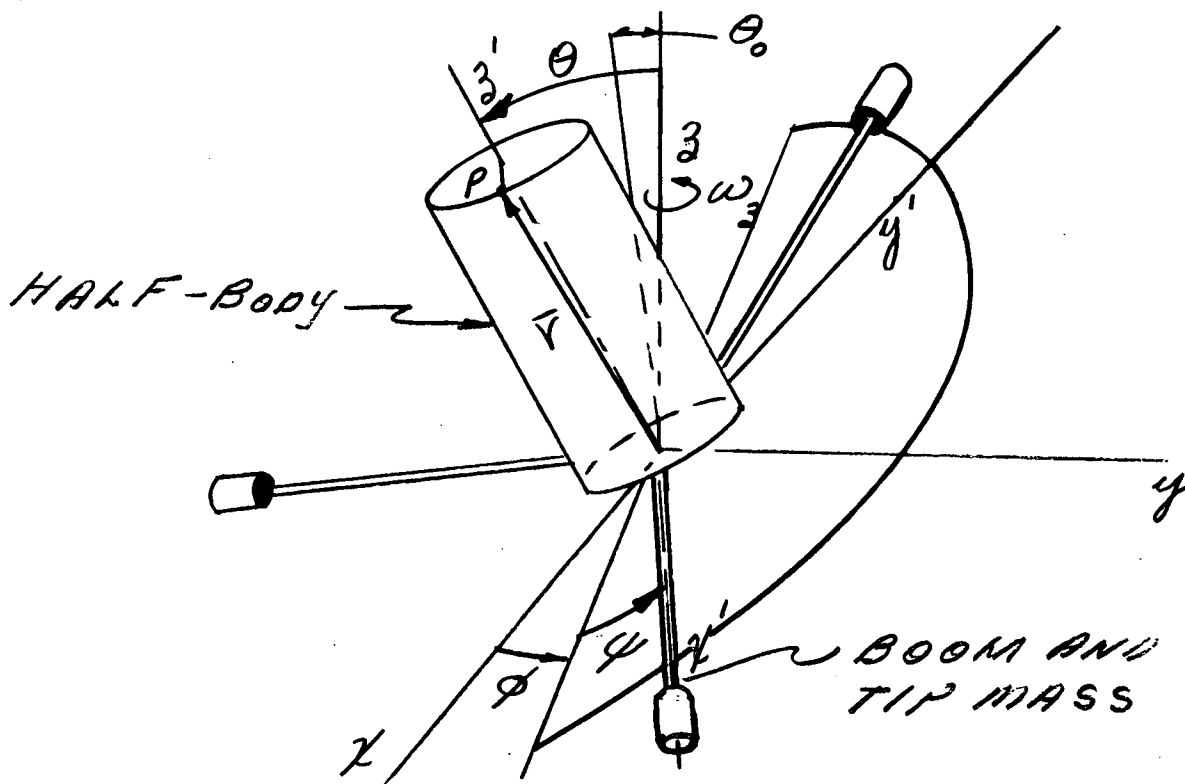


FIGURE E-4. - BOOM BENDING STUDY

for the moment at point i along the boom

$$\begin{aligned} \bar{M}_i = & \omega_{0z}^2 \frac{I_z}{I_x} \left(2 - \frac{I_z}{I_x} \right) \theta \sin \dot{\psi} t \times \\ & \left(I_{MB} - \frac{M_B L}{2} r_{xi} + \frac{M_{xi} r_{xi}^2}{6} \right) \bar{J} \end{aligned} \quad (2.22)$$

in which I_{MB} is the boom moment of inertia about $r_{xi} = 0$, i.e.,

$$I_{MB} = M_B \frac{L^2}{3}$$

L is the boom length and M_{xi} and r_{xi} are shown below. It is to be noted

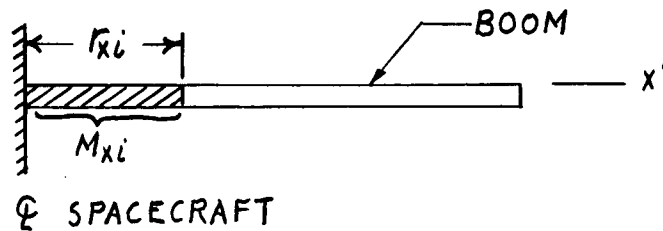


FIGURE E-5. - BOOM STUDY

that this moment is in a direction \bar{J} , normal to the x' -axis.

The bending moment \bar{M}_i , given by Equation (2.22) is periodic, with its period being found from the second of Equations (2.11). Thus, with θ small $\dot{\psi}$ is constant and

$$\dot{\psi} t_0 = 2\pi$$

or

$$t_0 = \frac{2\pi}{\omega_{0z} (1 - I_z/I_x)} \quad (2.23)$$

The bending stress is given by

$$\sigma = \frac{M_i s}{I_{y'}} \quad (2.24)$$

in which $I_{y'}$ is the second moment of the boom cross-sectional area at r_{xi} about the y' -axis.

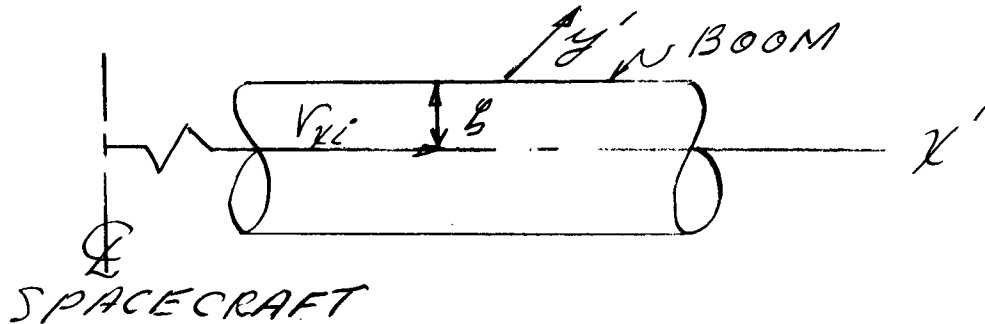


FIGURE E-6. - BOOM BENDING

We may now compute the drift rate $\dot{\theta}$ for the spacecraft resulting from boom hysteresis since we have obtained σ (Formula 2.24) and t_0 (Formula 2.23).

2.2.3 Body Bending. - We now account for hysteresis loss resulting from bending of the main body of the spacecraft. As before, we are only interested in the sinusoidal portion of the motion since it is only this which will cause bending (from inertia forces).

The bending moment at any location i along the spacecraft body is given by

$$\begin{aligned} \bar{M}_i = & \omega_{0z}^2 \left(\frac{I_z}{I_x} \right)^2 \theta \left[\left(\frac{M_R r_{zi} h}{4} + \frac{I_{yR}}{2} \right. \right. \\ & \left. \left. - \frac{M_{zi} r_{zi}^2}{6} \right) \sin \psi t \right] - \left(\frac{M_R r_{zi} h}{4} \right. \\ & \left. + \frac{I_{xR}}{2} - \frac{M_{zi} r_{zi}^2}{6} \right) \cos \psi t \quad (2.25) \end{aligned}$$

in which

M_R = Mass of the main body of the spacecraft

h = half length of the body.

The mass M_{zi} and r_{zi} are shown below and the other terms have been previously defined, except for

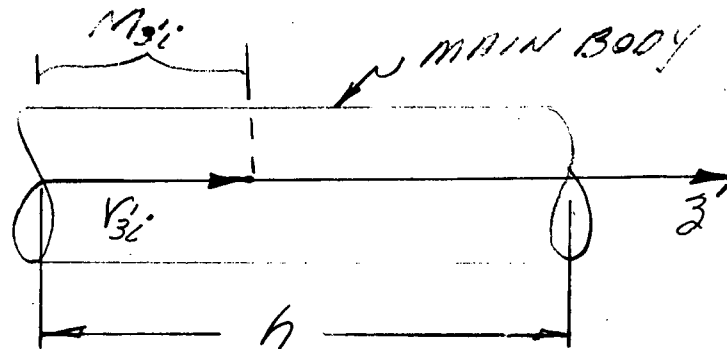


FIGURE E-7. - BODY BENDING

I_{XR} and I_{YR} which are:

I_{XR} = moment of inertia of the main body (no booms) about the x' -axis.

I_{YR} = moment of inertia of the main body (no booms) about the y' axis.

The bending stresses occur about both the y' and x' -axes. They are

$$\sigma_{by'} = \frac{M_{iy'} \eta_{ix'}}{I_{y'}} \quad (2.26)$$

$$\sigma_{bx'} = \frac{M_{ix'} \eta_{iy'}}{I_{x'}} \quad (2.27)$$

in which $I_{y'}$, $I_{x'}$, are the second moments of the shell (structural portion) in bending, about the y' and x' -axes, respectively. $\eta_{ix'}$, $\eta_{iy'}$, are the centroidal fiber distances along x' and y' respectively.

The period of bending in this case is again given by Equation (2.23).

2.2.4 Time Variation. - Through the use of Equation (2.21), in which σ and t_0 are given by Equations (2.23), (2.24), (2.26) and (2.27), we may write

$$\frac{\dot{\theta}}{\theta} = \frac{(E_R + E_B) I_z}{H^2 (I_z/I_x - 1)} \quad (2.28)$$

From which we obtain

$$\tau = \frac{1}{K} \log \left(\frac{\theta}{\theta_0} \right) \quad (2.29)$$

in which

$$K = \frac{(E_R + E_B) I_z}{H^2 (I_z/I_x - 1)} \quad (2.30)$$

and

$$H = I_z \omega_{0z} \quad (2.31)$$

E_R and E_B are the energy dissipated by the main body and booms, respectively, t is the time required to gain the cone half-angle θ_1 , and θ_0 is the initial steady state cone half-angle error. Two plots of cone angle growth versus time are shown in Figures E-8 and E-9, for various hysteresis damping factors, γ .

2.2.5 Conclusions. - It is concluded, on the basis of this section of the study that

- (1) The small allowable θ_i (Table 1), requires a control system with a small steady state error ($\ll .16^\circ$, for $\omega_z = 5.83$ radians/sec.)
- (2) The boom-extended configuration results in rapid cone build-up (in terms of hours).
- (3) The retracted boom configuration results in a very slow build-up of cone-motion (in terms of days).

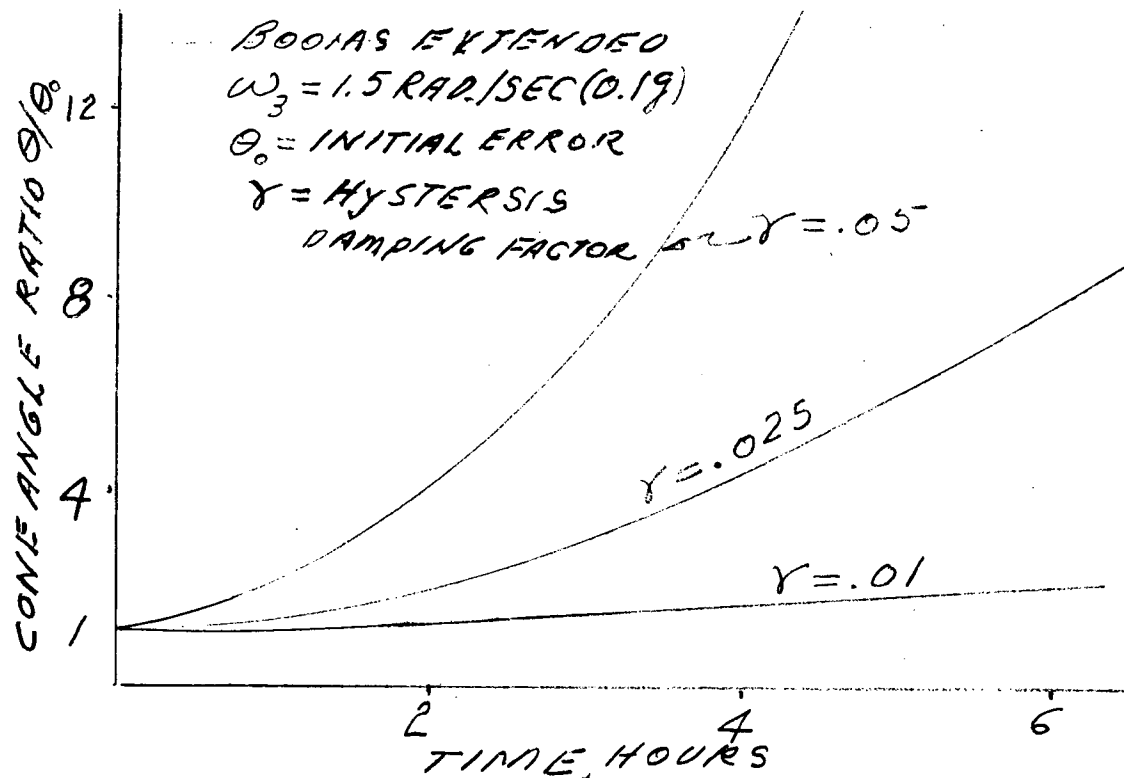


FIGURE E-8. - CONE ANGLE GROWTH

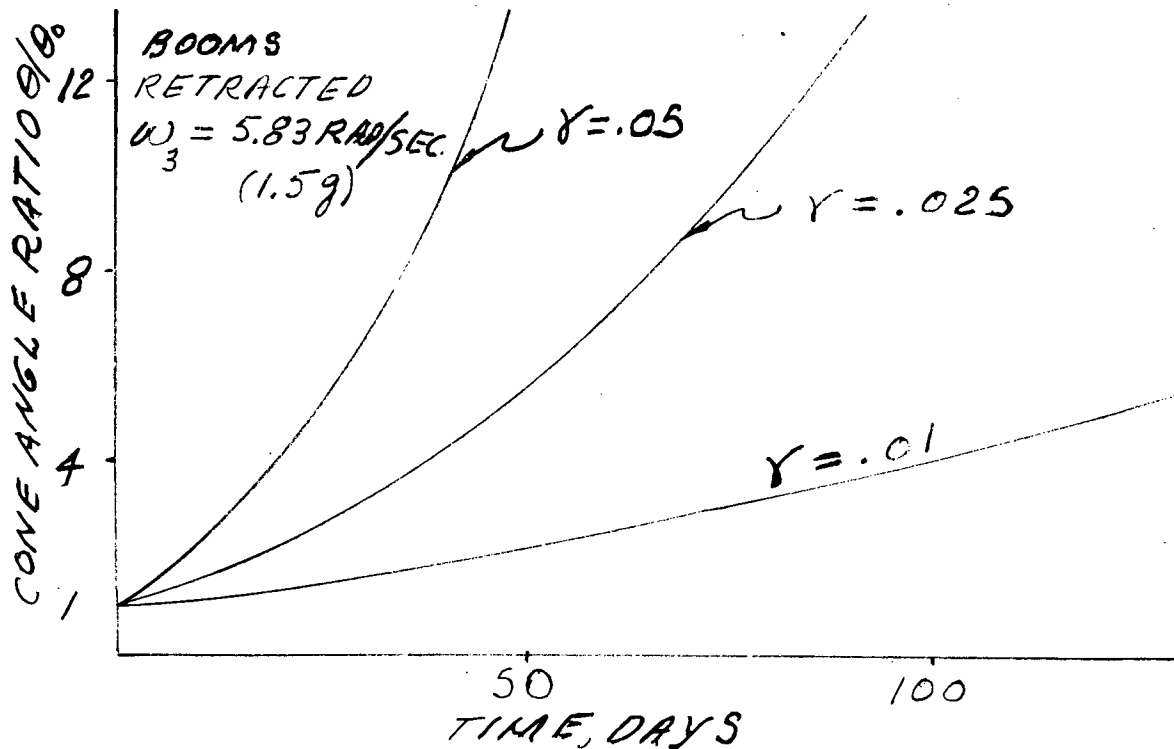


FIGURE E-9. - CONE ANGLE GROWTH

3.0 THERMAL BENDING

This section concerns itself briefly with the forced bending motion of the extended booms of the spacecraft as it spins in orbit about the earth, as shown in Figure E-10.

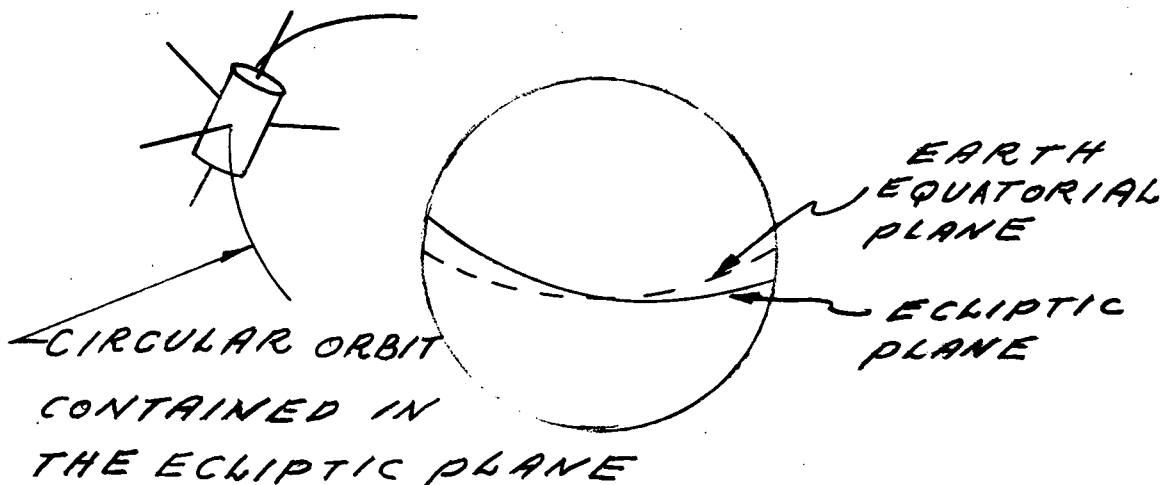


FIGURE E-10. - ORBITING SPACECRAFT

The spacecraft is in orbit about the earth but it is also rotating about its spin axis (z-axis) at the following spin rates in two different modes of operation.

$$\left. \begin{aligned} \omega_z &= 5.83 \text{ radians/sec (max.)} \\ \omega_z &= 1.50 \text{ radians/sec (min.)} \end{aligned} \right\} \quad (3.1)$$

The motion of the booms will include both the free vibratory phenomena, including possible dynamic instabilities and forced motion. The possibility

of forced resonance is investigated briefly in this section, while an investigation of dynamic instability is contained in Section 4.

We now look more closely at the spacecraft-sun relationship as indicated in Figure E-11.

We base our analysis on the work of Merrick (Reference E-1) and write

$$\dot{M}_{z'}(x, t) + \lambda M_{z'}(x, t) = -K \phi(x) \sin \beta t \quad (3.2)$$

in which λ is the reciprocal of the thermal time constant, M_z is the bending moment about the z' -axis, at any location x along the boom, $-k$ is a constant which depends upon the thermal, mechanical and geometric properties of the boom and ϕ is the angle of twist of the boom, relative to its root.

Now ϕ is taken to be a slowly varying function of time (as has been verified for this study) and hence treated as a parameter in this analysis.

If we now express the solution of Equation (3.2) in terms of principal coordinates, we have

$$M_y = - \sum_{i=1}^{\infty} EI V_i''(x) N_i(t) \quad (3.3)$$

in which V_i is a solution to the homogeneous equation

$$EI V_i''''(x) - m \omega_i^2 V_i(x) = 0 \quad (3.4)$$

in which E is Young's modulus, I is the second moment of the boom cross-section about the boom's centroidal axis along z' , m is the boom-mass per unit length, ω_i the i th natural frequency of the boom, and $N_i(t)$ is the i th generalized coordinate.

If we now substitute (3.3) into (3.2), multiply this result by V_i and integrate over the length of the boom, we obtain,

$$\dot{N}_i(t) + \lambda N_i = \bar{K}_i \sin \beta t \quad (3.5)$$

in which

$$\bar{K} = \frac{K \int_0^L \phi(x) V_i'' dx}{EI \int_0^L (V_i'')^2 dx} \quad (3.6)$$

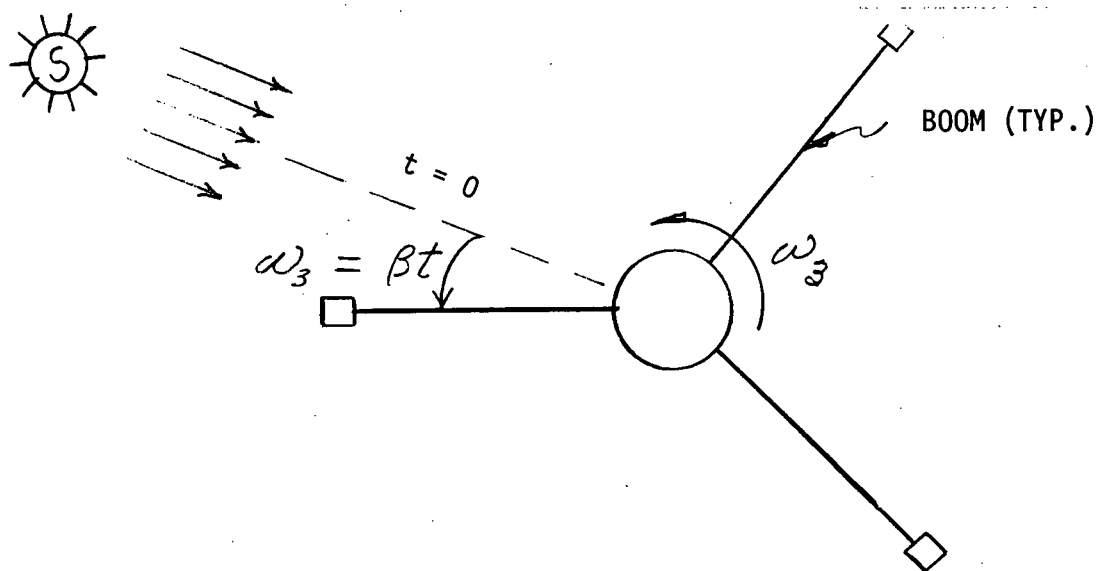


FIGURE E-11. SPACECRAFT-SUN RELATIONSHIP

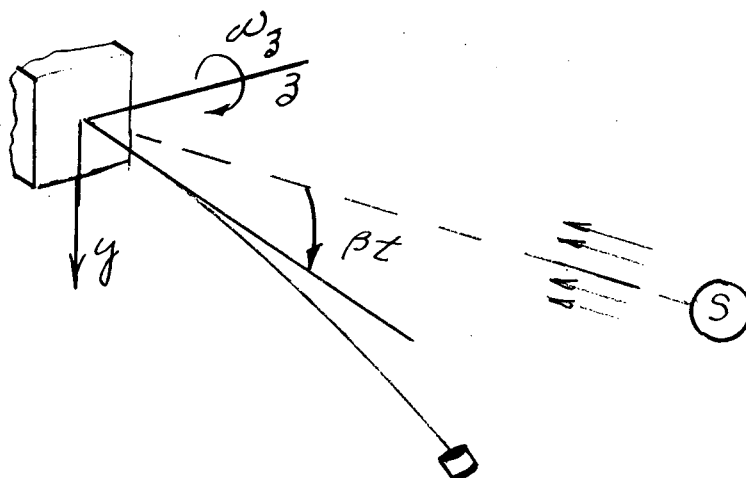


FIGURE E-12. BOOM BENDING

our approximation leads us to consider only the case $i = 1$ and we are to show that there is no danger of resonance as associated with Equation (3.5).

The solution to (3.5) is

$$N_i(t) = \left(\frac{K}{\lambda^2 + \omega_z^2} \right) [\sin(\omega_z t - C) + e^{-\lambda t} \sin C]$$

in which C is a phase angle. It is seen that M_z is a periodic function, with a frequency of ω_z .

The lowest boom natural frequency is approximated by

$$\omega_1^2 = \frac{3EI}{m\ell^3} \left[1 + \frac{m\rho\ell^2\omega_z^2}{2EI} \right] \quad (3.7)$$

Hence it is seen that

$$\omega_1 > \omega_z \quad (3.8)$$

so that there is never any danger of resonance.

4.0 THERMAL TORSION-BENDING DYNAMIC STABILITY

We now look at the possibility of instabilities that might occur as a result of the interaction of the temperature-induced moments and torques and the coupled bending-torsion response of a typical boom.

4.1 Sun Time and Location. - It can be seen from Figure E-11 that if we take $t = 0$ to occur at the time when Boom A aligns itself with the sun's rays, the times for heating a specific side of a boom are as follows:

TABLE 2

<u>SPIN SPEED</u>	<u>TIME/CYCLE</u>	<u>SUNLIGHT TIME, t_s</u>
5.83 radians/sec	1.08 sec.	.54 sec.
1.50 radians/sec.	4.18 sec.	2.09 sec.

The incident radiation during t_s which is absorbed by a boom results in boom bending. This boom bending is generally coupled to a torsional response (this is particularly true for sections which are open torsionally) and the final result is a self-excited vibration not unlike the concept of aerodynamic flutter or the "shimmy" of an aircraft landing gear.

The basic ingredient which governs the dynamic stability of a boom is the torque that is built up in a section governed by the derivative of $V(\psi_0)$, (Reference E-2), in which ψ_0 is the angle of the sun relative to the boom cross-section. If the derivative

$$\frac{\partial V}{\partial \psi_0} < 0 \quad (4.1)$$

the boom will tend toward instability and if

$$\frac{\partial V}{\partial \psi_0} > 0 \quad (4.2)$$

the system will tend toward stability.

A plot of the function

$$\frac{\partial V}{\partial \psi_0}$$

is taken from Reference E-2 and reproduced in Figure E-13.

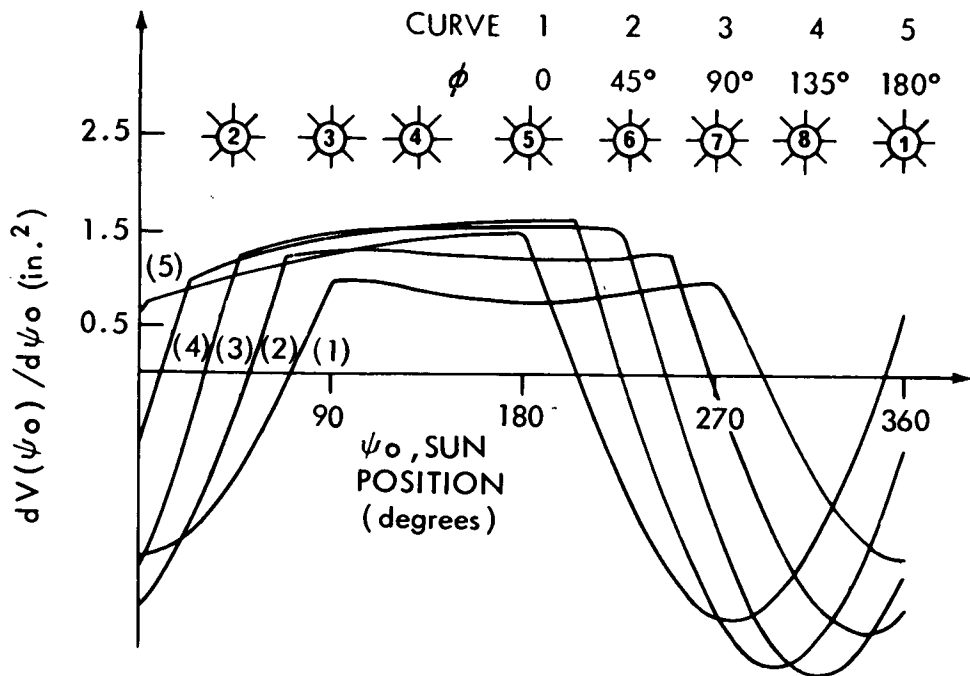


FIGURE E-13. - DERIVATIVE OF $V(\psi_0)$

On the basis of the above, it is clear that we may avoid instability which takes place in the region of

$$\frac{\partial V(\psi_0)}{\partial \psi_0} < 0$$

by the proper orientation of the booms relative to the spacecraft main-body. These suggested orientations are shown in Table 3 in conjunction with Figure E-14.

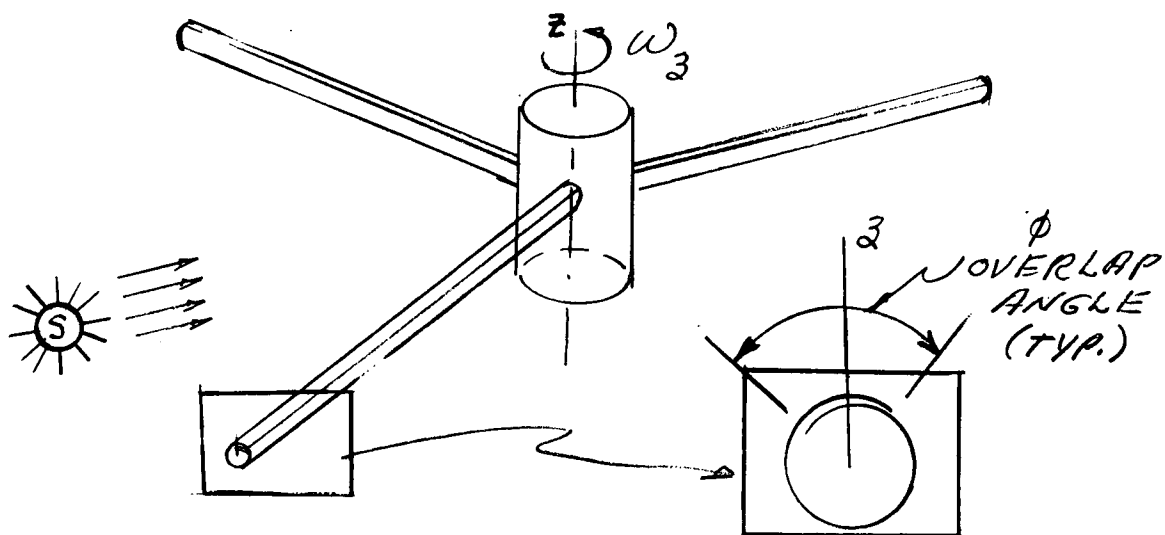


FIGURE E-14. - BODY-BOOM SYSTEM

TABLE 3. - SUGGESTED BOOM ORIENTATION

SYSTEM	1	2	3	4	5
OVERLAP ANGLE	0°	45°	90°	135°	180°
REGIONS OF STABILITY ($2V > 0$)					
SUGGESTED ORIENTATION					

A further analysis was carried out, based on the work of Merrick (Reference E-1). On this basis, it was found that the system could tend toward instability when thermal loading was applied so that the negative derivatives of $V(\psi_0)$ were applicable. It was found on this basis that the system tends to slight instability but amplitude build-up is possible over periods of sunlight that are much greater than the actual operating case. In fact we have the following.

MAXIMUM
TIME IN SUNLIGHT
PER REVOLUTION

$$t_s = 2.09 \text{ sec.}$$

TIME FOR
UNSTABLE DOUBLE
AMPLITUDE OF
MOTION

$$35 \text{ sec.}$$

It was found that the amplitude growth increased as follows:
If the amplitude is A_0 at time $t = 0$, the amplitude will grow to, in time t_s ,

$$A_0 e^{.02(2.09)} = A_0 (1.0418)$$

Thus, during the time t_s , the amplitude will grow by only 4%.

The above ideas were based on the concept that within t_s the system is subjected to full radiation. Of course, this is highly conservative and the tendency toward instability is even less than indicated above.

5.0 SPACECRAFT CONTROL RESPONSE

A parametric study was performed on the stability of a typical spinning satellite having a rigid central body with flexible booms. The effect of boom length on rate of divergence of wobble amplitude was found. From these results the rate of reaction control propellant consumption required to maintain this amplitude within prescribed limits was estimated assuming discrete corrections.

5.1 PARAMETRIC STABILITY STUDY

A mathematical model of the spinning body with flexible booms was described using state variables and linearized equations of motion. Four degrees of freedom were assumed. These were:

Central body pitch and yaw angular rates

Assymmetric boom deflections, pitch and yaw planes

The eigen values or roots of this 6th order system were found as a function of three parameters which apply equally well to 3 or 4 boom configurations:

Central body spin/yaw inertia ratio

Ratio of boom to central body yaw inertias.

Boom stiffness parameter $f = \left(\frac{\omega_n}{\omega_{spin}} \right)^2$

The real and imaginary parts of the eigen values, $\sigma \pm j\omega$, were non-dimensionalized and presented as

ω / ω_{spin} and $\sigma / c/m$ where:

C = Effective dashpot at boom tip

m = Tip mass.

ω_n = Boom natural frequency stiffened by spin rate,

ω_{spin} = spin rate (held constant)

For typically small boom dashpot values the real part of the root was found to be directly proportional to the dashpot constant used. This justified expressing the desired results of this study, divergence rate, in terms of the non-dimensional parameter $\sigma / c/m$

The results of a systematic parametric variation on these eigen values are shown in Figure E-15. The main points found here are:

SYSTEM ROOTS
 $S = \sigma \pm j\omega$ RAD/SEC

ω_n = BOOM NAT. FREQ
 STIFFENED BY
 SPIN RATE

$$\frac{\omega}{\omega_{SPIN}}$$

m = TIP MASS

C = EFFECTIVE DASHPOT
 AT BOOM TIP

$$\left(\frac{I_{SPIN}}{I_{PITCH}} \right) = 0.4$$

$$\frac{\sigma m}{C}$$

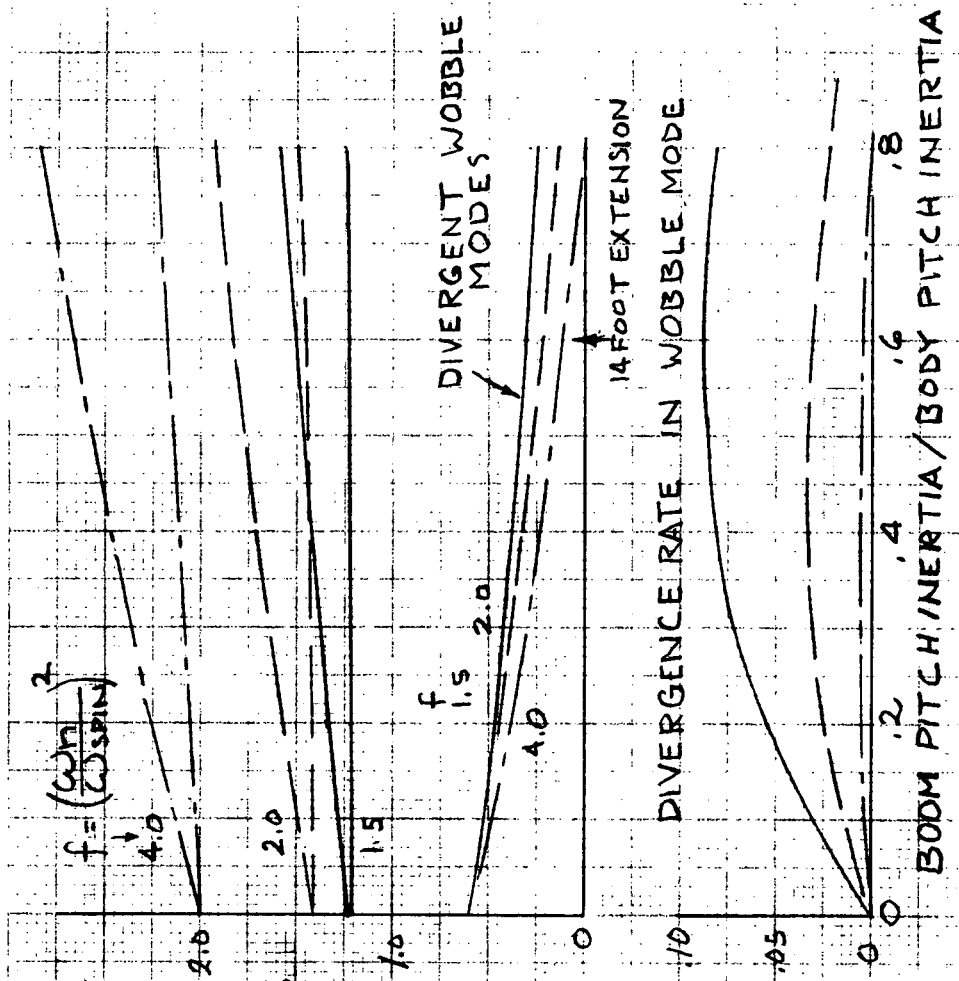


FIGURE E-15. - BOOM INERTIA AND STIFFNESS PARAMETRIC STABILITY STUDY

- Flexibility increases boom inertia required to attain neutral stability. Thus:

$f = \left(\frac{\omega_n}{\omega_{spin}} \right)^2$	RIGID	4.0	2.0	1.5
$\frac{\Delta I_{spin boom}}{I_{spin body}}$.6	.8	1.2	1.8

- Maximum divergence rate at intermediate boom length is a strong function of stiffness.
- Higher frequency modes are damped.

5.1.1 Baseline Configuration Characteristics. - The variation of the inertia and stiffness parameters as booms are extended was estimated for one typical Bioresearch Module configuration. Figure E-16 shows boom natural frequency versus boom extension. The spin rate and overall vehicle spin-to-pitch inertia ratio for the assumed vehicle is also shown. These two parameters are independent of boom flexibility. The effective spring rate of the isolated cantilever boom drops rapidly with length as evidenced by the rapid decrease in natural frequency shown without spin. The isolated half inch STEM boom has a natural frequency less than the spin rate for practically all extensions. The presence of spin provides restoring forces at the tip mass which effectively stiffen the booms. The boom frequency as stiffened by spin is 35 to 40% above the spin rate for extensions over 50%.

The boom natural frequency was calculated from the following expression:

$$\omega_n^2 = \frac{3EI}{ml^3} + \frac{3}{2} \left(1 + \frac{r_0}{l} \right) \omega_{spin}^2 \quad (5.1)$$

where:

$$EI = 1800 \text{ lb-in.}^2$$

$$m = 1.33 \text{ lbs tip mass}$$

$$r_0 = 1 \text{ foot, boom attachment radius}$$

$$l = \text{Boom length}$$

5.2.2 Divergence Time. - The parameters for the particular extending boom configuration of Figure E-16 were combined with the parametric study of Figure E-15 and data on STEM boom damping measurements, to obtain the divergence times shown in Figure E-17.

Damping was obtained from the experimental data on a STEM

ω_n = BOOM NAT. FREQ.
STIFFENED BY SPIN
RATE

$\frac{I_{SPIN}}{I_{PITCH}}$ = MOMENT OF
INERTIA
RATIO
ASSUMING RIGID
BOOMS

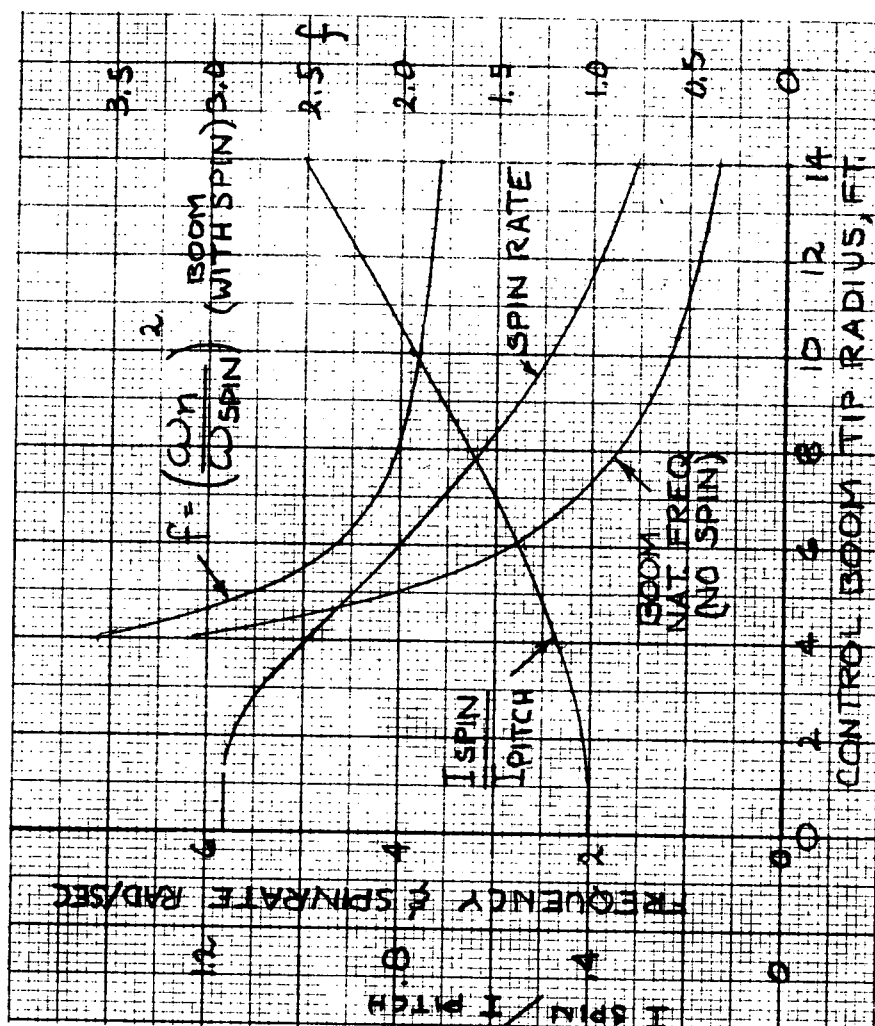


FIGURE E-16. - CONTROL BOOM INERTIA AND STIFFNESS CHARACTERISTICS

$1/\sigma$ = TIME TO
 DIVERGE TO $e \times$
 INITIAL WOBBLE
 AMPLITUDE

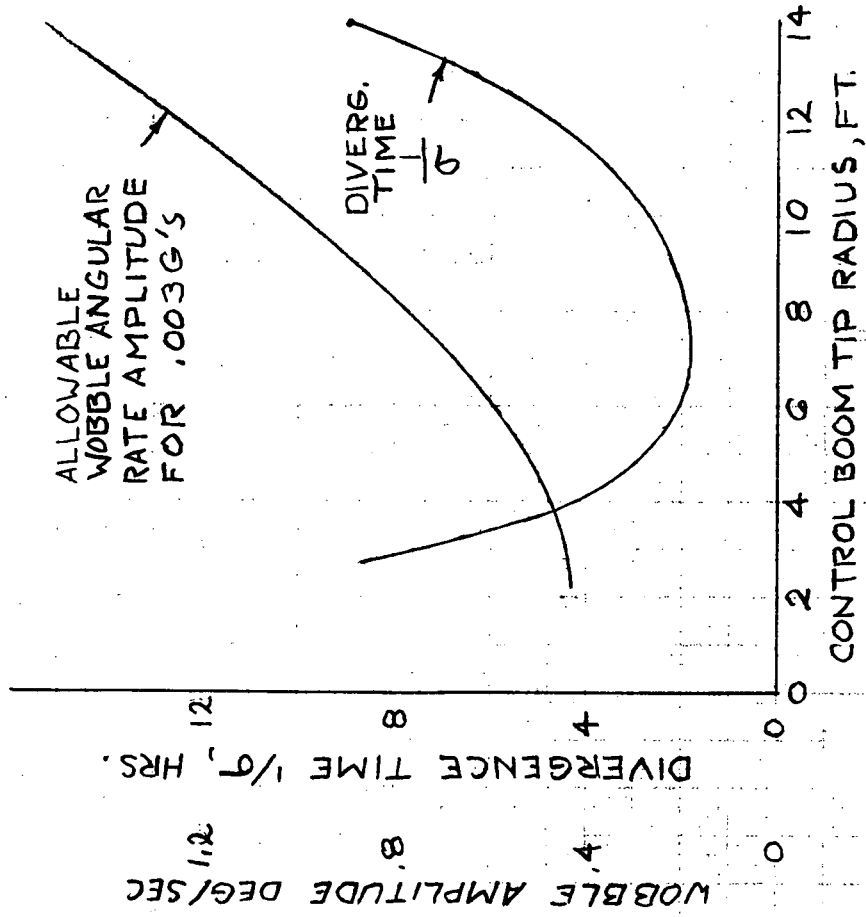


FIGURE E-17. - WOBBLE DIVERGENCE TIME AND ALLOWABLE AMPLITUDE

boom cantilevered vertically under the influence of gravity as reported in Reference E-3. These tests indicated that the typical log decrement, δ , was 0.01 for oscillations of a 43 foot boom. Variations in test conditions caused this parameter to vary by a factor of two. The test natural frequency was estimated and the hysteresis factor

$$\gamma = c/k = \delta/\pi\omega$$

was found to be about .003 for two different boom lengths. It was then assumed that this factor did not change with addition of tip masses.

The factor C/m was estimated for the satellite configuration with boom tip mass added from:

$$c/m = \frac{c}{k} \cdot \frac{k}{m} = \omega_{n_0}^2 \left(\frac{c}{k} \right) = .003 \omega_{n_0}^2 \quad (5.2)$$

$$\text{where } \omega_{n_0}^2 = \frac{3EI}{ml^3}$$

is the natural frequency of the isolated cantilever boom with tip mass, m , added.

Finally the divergence rate, σ , was obtained from the non-dimensional damping parameter,

$$\frac{\sigma}{c/m}, \text{ thus,}$$

$$\sigma = \frac{c}{m\omega_z} \left(\frac{\sigma}{c/m} \right) = .003 \omega_{n_0}^2 \cdot \left(\frac{\sigma}{c/m} \right) \quad (5.3)$$

The time for wobble amplitude to increase exponentially by a factor of 2.72 is given by $1/\sigma$ as shown in Figure E-17. This time constant varies from 1.8 hours at 5 foot extension to 9 hours at full extension. The control system must provide discrete corrections before rate gyro amplitude reaches the allowable rate values shown in Figure E-17. Variation in acceleration at the forward end of the experiment is held below 3×10^{-3} g by this requirement.

5.3 PROPELLANT CONSUMPTION

For the purpose of estimating reaction control propellant consumption it was assumed that a rate gyro would be activated for brief periods during amplitude buildup. No correction would be made until the sampled rate exceeded a prescribed value. At that time control pulses would be applied until sensed rate dropped below a prescribed minimum value, at which time the correction would cease, starting a new buildup cycle. The time between corrections is given by

$$\Delta t = \frac{1}{\sigma} \log \left(\frac{a_2}{a_1} \right)$$

where a_2/a_1 is the ratio of the rate amplitudes before and after correction. Propellant consumption could be minimized by setting the maximum rate as small as possible but a limit is reached when the minimum rate is less than obtainable rate gyro accuracy.

Figure E-18 shows a typical response for the chosen compromise settings. The correction was assumed to occur from 0.4 to 0.1 deg/sec which would provide 12.5 hrs. between pulses for the full extension condition where $1/\omega$ is 9 hours.

The impulse required to make this 0.3 deg/sec correction is .077 lb-sec assuming the control jet acts on a 1.5 foot arm and the transverse moment of inertia is 22 lb-ft-sec².

The reaction control impulse expended per day was computed as a function of boom length on the basis of the above assumptions. Figure E-18 shows that consumption varies from .155 lb sec/day at full extension and very small values with full retraction to a maximum of about .75 lb-sec/day when the experiment is maintained at 0.6 g. This is equivalent to 2.25 lbs of N₂ consumed in 6 months.

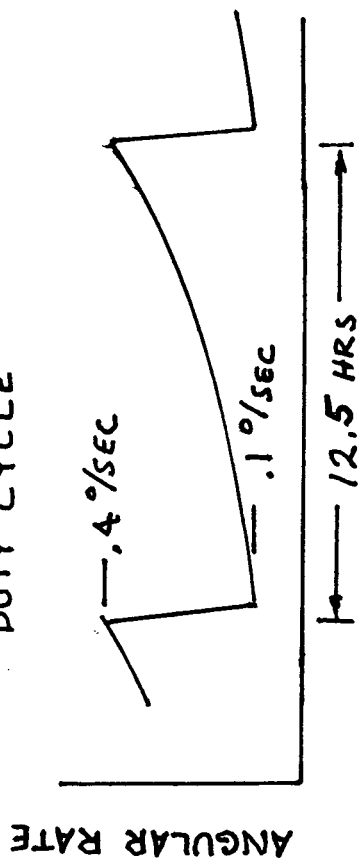
5.4 RATE GYRO RESPONSE TO FLEXIBLE BOOMS

When control motors are pulsed the flexible booms will be excited superimposing an unwanted oscillation on the transverse body rates sensed by the body mounted rate gyros. The transient response of the sixth order system used in the stability analysis was calculated at full-extension of the booms. Figure E-19 shows the variation in the transverse body rate vector produced by releasing initially restrained tip masses while the central body is coning. This vector would trace out a circle, if the booms were rigid. The flexible booms cause a bending oscillation at about 8 times the coning frequency or 1.8 times the spin frequency. The amplitude of this motion is only about 15% of the total transverse rate amplitude indicating that the rate gyros will function essentially as in the rigid body case.

5.5 CONCLUSIONS

The results of this study showed that flexible booms cause the spinning module cone angle to increase. With booms retracted or fully extended the growth is very slow. Growth rate is most rapid at the intermediate boom length required for 0.6 experiment g. The propellant required to compensate this divergence for 6 months is 2.25 lbs of nitrogen. The boom bending frequency component was found to have little effect on rate gyro measurement of body rates for wobble damping control.

TYPICAL
WOBBLE AMPLITUDE
DUTY CYCLE



14 FOOT EXTENSION
OR 0.1 EXPERIMENT "G's"

PROPELLANT IMPULSE =
0.155 LB·SEC / DAY
28 LB·SEC IN 180 DAYS

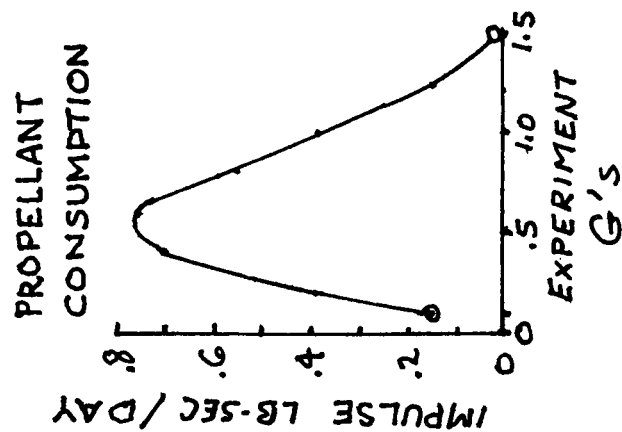


FIGURE E-18. - WOBBLE CONTROL DUTY CYCLE AND PROPELLANT CONSUMPTION

BODY RATE VECTOR INITIAL MOTION IN BODY AXIS SYSTEM

INITIAL CONDITIONS:
 X-AXIS BODY RATE
 TIP MASSES
 RESTRAINED

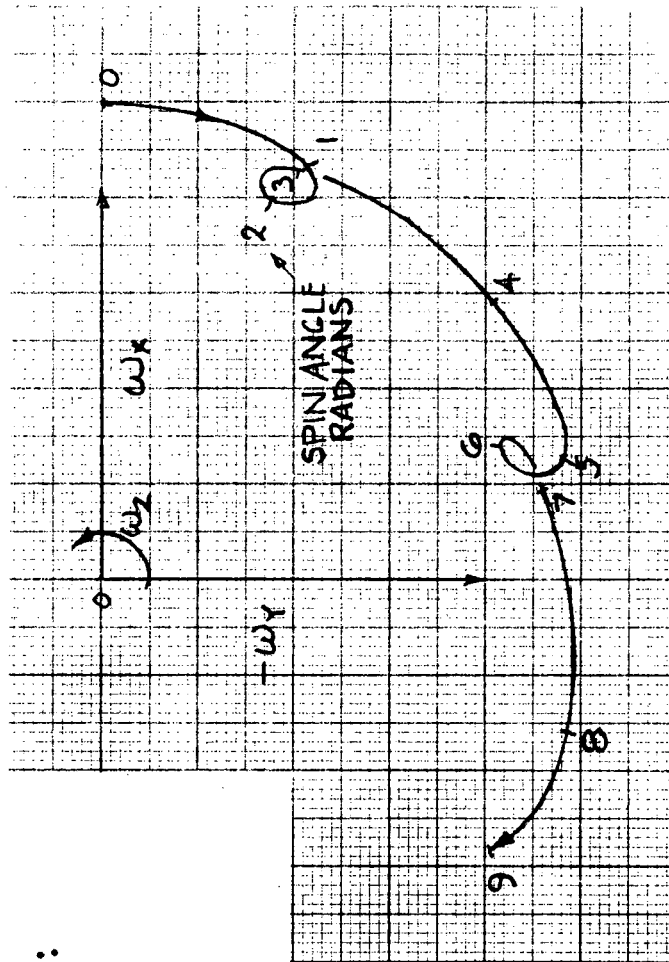


FIGURE E-19. - TYPICAL BODY RATE INITIAL RESPONSE

REFERENCES

- E-1. Vernon K. Merrick, "Instability of Slender Thin-walled Booms Due to Thermally Induced Bending Moments," NASA TN D-5774, May, 1970.
- E-2. Harold P. Frisch, "Thermal Bending Plus Twist of a Thin-Walled Cylinder of Open Section with Application to Gravity Gradient Booms," NASA TN D-4069, August 1967.
- E-3. R. E. Predmore, C. L. Staugaitis, and J. E. Jellison, "Damping Behavior of Dehavilland Stem Booms," NASA TN D-3996, June, 1967.

APPENDIX F

BIORESEARCH MODULE PROGRAM MILESTONES AND WORK FLOW

This appendix contains four parts: a listing and description of potential Contract Line Items; a Work Breakdown Structure that diagrams the structure to the third level and then indexes the structure to the sixth level or indenture; a Program Plan or work flow; and a Schedule of major program milestones.

1.0 CONTRACT LINE ITEMS

Potential Contract Line Items are identified below accompanied by brief description of their contents.

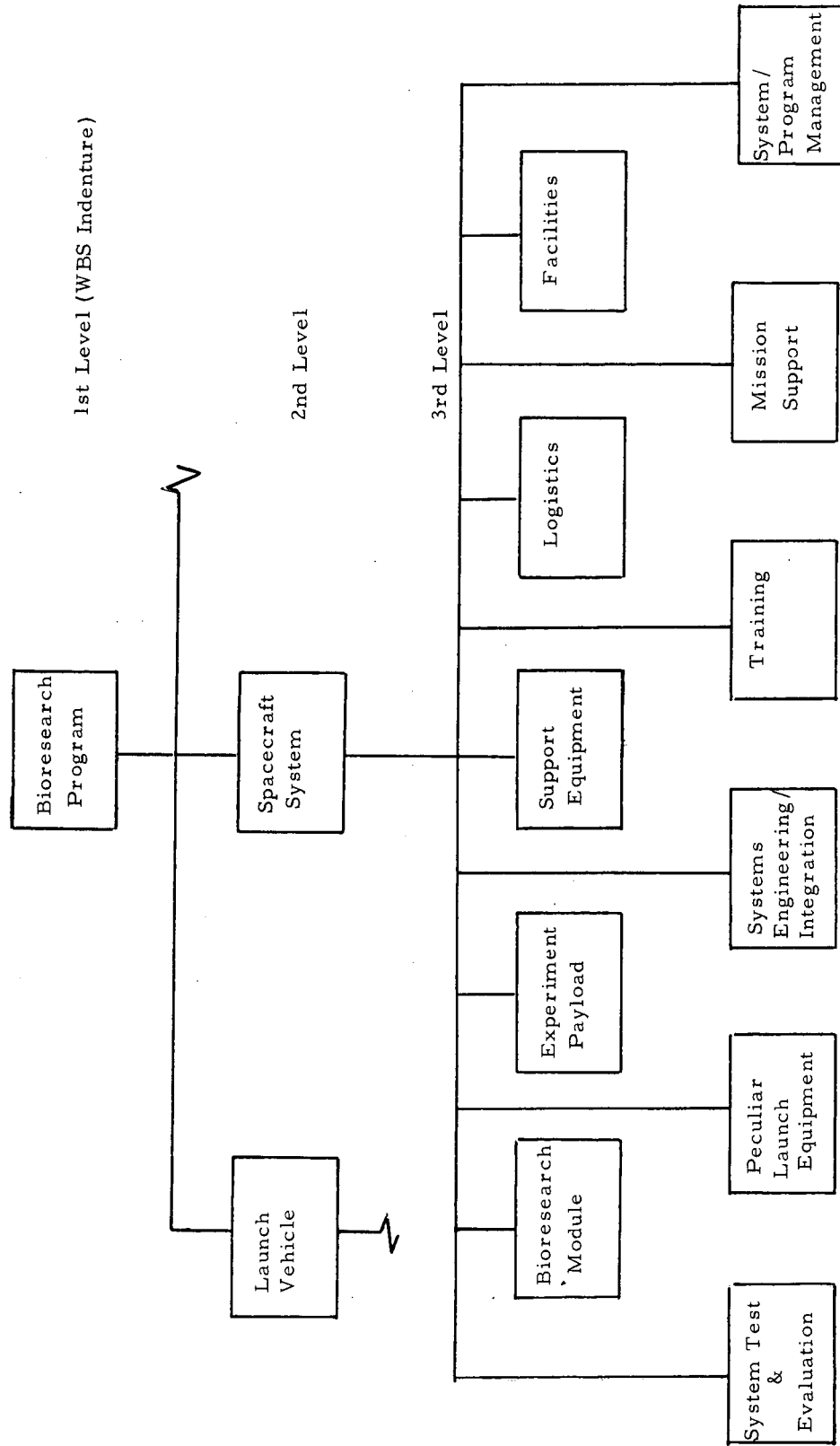
<u>CLI No.</u>	<u>Description</u>
1.	Bioresearch Modules; four (4).
2.	Bioresearch Module and System launch, checkout and servicing Ground Support Equipment.
3.	Experiment Life Support and Handling / Servicing Ground Support Equipment.
4.	Requirements / Design Package for Bioresearch Module System (including experiments) orbital and mission operations and orbital/mission peculiar Ground Support Equipment.
5.	Procedures / Instructions for handling, servicing, inserting, and withdrawal of experiment(s) using the Experiment Life Support and Handling / Servicing GSE and maintenance of the procedures / instructions.
6.	Experiment / Bioresearch Module System Technical and Operations Services for Orbital Missions; 4 months each for 4 missions.

<u>CLI No.</u>	<u>Description</u>
7.	Reliability and Quality Assurance for Bio-research Modules, System GSE, and Experiment and Life Support GSE.
8.	Bioresearch Module System checkout, and booster integration services.
9.	Bioresearch Module System Engineering.
10.	Mission Integration and Trajectory Analyses.
11.	Bioresearch Module Structural Prototype.
12.	Program/System Logistical Support (Inventory control, stock, store, and issue).
13.	Bioresearch Module System Integration and Acceptance Testing.
14.	Bioresearch Module System Program Management and Integration.

2.0 WORK BREAKDOWN STRUCTURE

The Work Breakdown Structure appears in Table F-1 in diagrammatic form to the third level and also in indentured index form to the sixth level.

TABLE F 1 WORK BREAKDOWN STRUCTURE DIAGRAM



WBS INDENTURES

1	2	3	4	5	6
	BIO RESEARCH PROGRAM				
	Spacecraft System				
	Bioresearch			Module	
	Attitude Control Subsystem				
	Control Electronics				
	Nitrogen Tanks				
	Nitrogen Regulator				
	Thruster Valves				
	Yo-Yo Assembly				
	Sun Sensors				
	Rate Gyro Assembly				
	Integrating Rate Gyro				
	Horizon Crossing Indicators				
	Extendible Booms				
	Thermal Control Subsystem				
	Louver Assemblies				
	Radiating Cold Plate Assemblies				
	Thermistor Assemblies				
	Thermal Control Electronics				
	Insulating Blankets				
	Cold Plates				

1	2	3	4	5	6
					Communications and Telemetry Subsystem
					Command Receiver Assembly
					Command Decoders
					Programmer Clock
					Signal Conditioner
					Data Processor
					PCM Encoder
					Telemetry Transmitters
					Data Storage Assembly
					N ₂ Press Transducer
					Turnstile Antenna
					Antenna Coupler
					Tracking Beacon
					Data Patch Unit
					Dipole Antenna
					Range-Range Rate Transponder
					Harness
					Electrical Subsystem
					Power Control Assembly
					Solar Cells
					Battery Assembly
					Power Distribution Box
					Harness
					Structure
					Experiment Package Cover
					Equipment Section
					Aft Section
					Scout Support Ring
					Umbilical Receptacle/Plate
					Extendible Solar Panels and Assembly

1	2	3	4	5	6
			Experiment Payload(s)		
			Mechanical Support Equipment		
			Trailer Van		
			Work Stand		
			Platform Scales		
			Roll-Over Stand		
			Hoist Sling		
			Spacecraft Simulator (Mechanical Fit Check)		
			Fluids and Pneumatics Support Equipment		
			Environmental Control Console		
			Thermal Exchange Unit		
			Experiment(s) - GFE		
			Experiment Canister(s) - GFE		
			Experiment Simulator(s) - GFE		
			For Attachment to -41 Cold Plate		
			For Attachment to -13 Cold Plate		
			Support Equipment		
			Mechanical		
			Shipping Containers		
			Hoist Crane		
			Spacecraft Hoist Sling		
			Work Stand		
			Roll-Over Stand		
			Scales		
			Solar Panel Covers		
			Experiment Pack Simulation (see Experiment Payload)		

Flight Control Console

Console	Console Harness	Flight Control Equipment Panel
Power Supply	Blowers	
DC Digital Voltmeter	Signal Generator	
E-Put Counter	Oscilloscope	
Servo Analyzer	Rate Table	
Megohmmeter	Power Control Panel	

Cables

1	2	3	4	5	6
					Spacecraft Mission Control - Electronic
					Mission Control Console
					Console
					Console Harness
					Command Control Panel
					Power Supply
					Blowers
					Signal Generator (100-1700 MHz)
					Frequency Meter
					Power Meter
					RF Load Meter
					Electronic Counter
					Function Generator
					Battery Charger
					Battery Load Tester
					Power Control Console
					Console
					Harness
					Power Supply
					Blowers
					Power Panel
					Experiment Control Console
					Event Control Panel
					Event Monitor Panel
					Cables

1						
2						
3						
4	Logistics					
5		Procurement/Scheduling				
6		Property Administration and Control				
		Inventory Maintenance				
		Vendor Experiment Liaison				
	Training					
		Course Requirements				
		Instruction				
	Facilities					
		Spacecraft Check Out				
		System Acceptance				
		Receiving/Inspection				
		Special Purpose Test				
		Experiment Build-Up				
		Spin Balance				
		S/C Check Out				
		Experiment Build-Up (GFE)				
		Ordnance Storage				
		Launch				
		Mission				
	System Test and Evaluation (GSE to be added)					
	Development					
		Electrical Performance				
		Data Processor Bread Board (-11)				
		Attitude Control Electronics Bread Board (-2)				
		Electrical and Mechanical				

1	2	3	4	5	6
					Electrical Performance
					Control Electronics Prototype (-2)
					Data Processor Prototype (-11)
					Thermal Control (-12) (-14) (-15) (-16) (-13)
					Programmer Prototype (-20)
					Power Controller Prototype (-30)
					Battery Assembly Prototype (-32)
					Balance and Spin Test (Prototypes) (-28) (-34) (-35)
					(-36) (-37 Ring plus Dummy Hardware)
					Physical Measurements
					Prototype Structure (-28) (-34) (-35)
					(-36) (-37 Ring)
					Temperature Humidity
					Programmer Clock Prototype (-20)
					Power Controller Prototype (-30)
					Data Processor Prototype (-11)
					Control Electronics Prototype (-2)
					Louver Control Electronics
					Prototype (-16)
					Battery Assembly Prototype (-32)
					Vibration/Shock
					Prototype Structure (-28) (-34) (-35)
					(-36) (-37 Ring Dummy Hardware,
					some (-31) and Simulated Cells

1						
2						
3						
4						
5						
6						Control Electronics Prototype (-2) Data Processor Prototype (-11) Louver Control Electronics (-16) Programmer Clock Prototype (-20) Power Control Assembly (-30) Battery Assembly Prototype (-32)
						Static Load Structural Design Verification Test
						Prototype Structure (-28) (-34) (-35) (-36) and (-37 Ring)
						Acceleration
						Control Electronics Prototype (-2) Data Processor Prototype (-11) Louver Control Electronics (-16) Programmer Clock Prototype (-20) Power Control Assembly (-30) Battery Assembly Prototype (-32)
						Thermal Vacuum Includes Calibration Thermal Control Subsystems -12, -41, -15, -16, -17, -34, -31, & -14

1						
2						
3						
4						
5						
6						
						Thermal Control Subsystem -12, -13, -14, -15, -16, -17, -34 and -31, Experiment Pack Simulator
						Antenna System(s) Qualification
						Acceptance
						Leak Detection
						N ₂ Tankage (-3)
						N ₂ Regulator (-4)
						Thruster (-5)
						Attitude Control Valves and Plumbing (-6)
						Ground Cooling System (Plumbing, Valving, (-38, -13/-41 Dummy)
						Radiating Cold Plate Assembly (-13)
						Cold Plate (-41)
						Electrical Performance
						Control Electronics (-2)
						Thrusters (-5)
						Sun Sensors (-9)
						Rate Gyros (-10)
						Integrating Rate Gyros (-40)
						Extendible Booms (-43)
						Horizon Crossing Indicator (-44)
						Thermistor Assembly (-14)
						Louver Control Actuator (-15)
						Louver Control Electronics (-16)

1 2 3 4 5 6

Thermal Control Fluid
Lines, Valves (-42)
Command Receivers (-18)
Command Decoders (-19)
Programmer Clocks (-20)
(Flight)
Signal Conditioning (-21)
Flight Data Processor (-11)
PCM Encoder (-22)
Telemetry Transmitters (-23)
Data Storage Assembly (-24)
Turnstile Antenna (-26)
Antenna Coupler (-27)
Data Patch Unit (-29)
Dipole Antenna (-45)
Dipole Antenna (-46)
Range Rate Transponder (-47)
Power Control Assembly (-30)
Solar Cells Panels (-31)
Flight Battery Assembly (-32)
Power Patch Unit (-33)
Umbilical Connector /
Receptacle (-38)

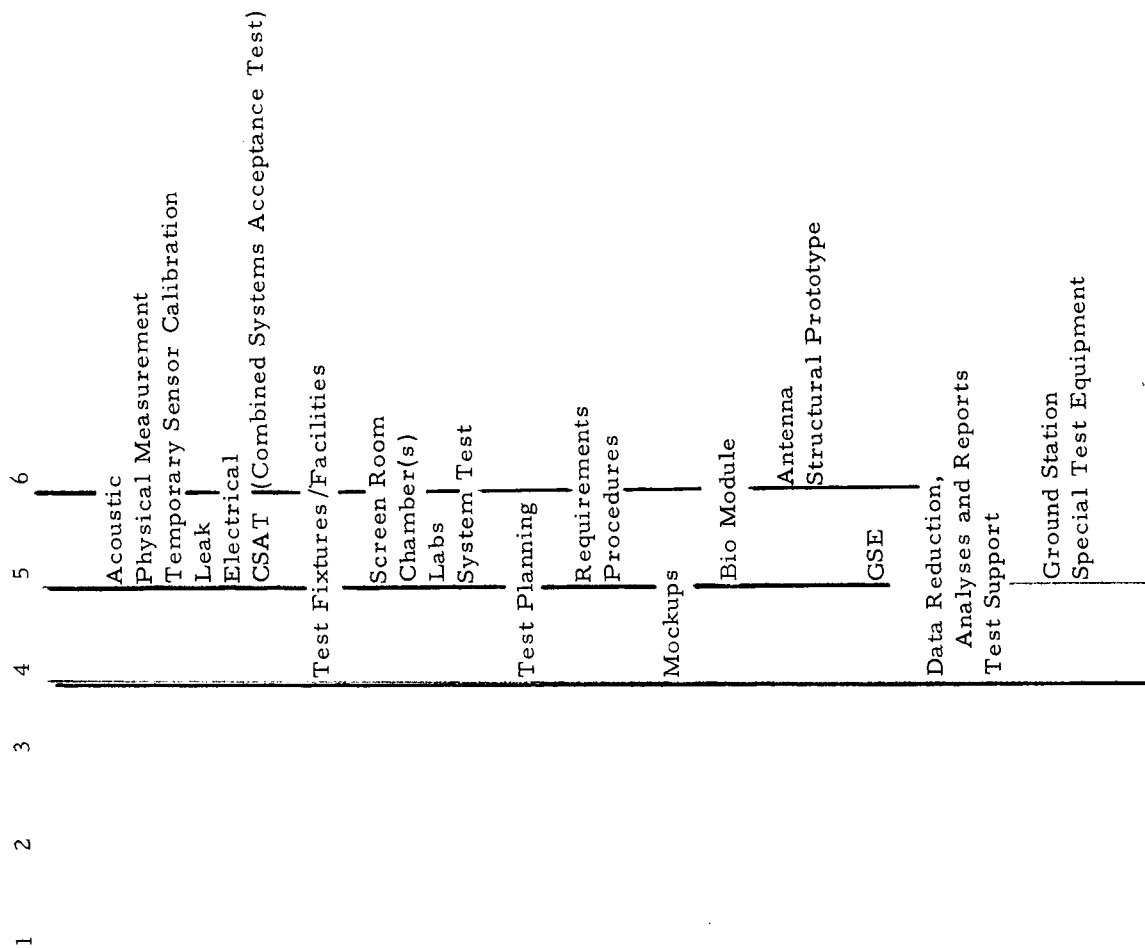
Physical Measurements

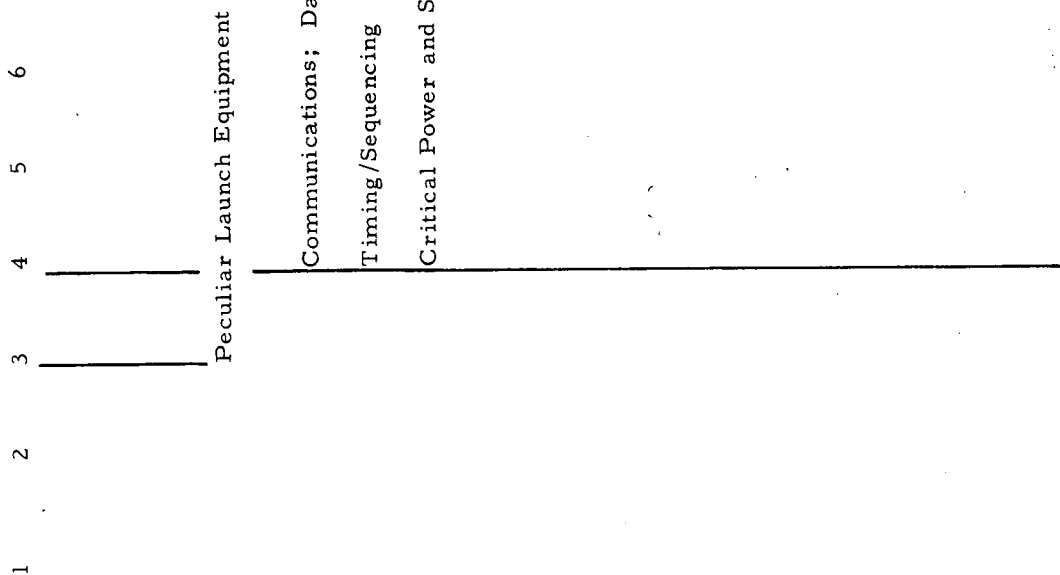
Control Electronics (-2)
N₂ Tanks (-3)
N₂ Regulators (-4)
Thrusters (-5)
Attitude Control Valves
and Plumbing (-6)
Sun Sensors (-9)
Rate Gyros (-10)
Yo-Yo Assembly (-39)
Integrating Rate Gyros (-40)
Extendible Booms (-43)

1 2 3 4 5 6

Horizon Crossing
Indicators (-44)
Thermal Control Sub-
systems Louver
Assembly (-12)
Radiating Cold Plate
Assembly (-13)
Thermistor Assembly (-14)
Louver Control Actuators
(-15)
Louver Control
Electronics (-16)
Insulation Blankets (-17)
Cold Plate (-41)
Thermal Control Fluid
Lines, Valves (-42)
Command Receivers (-18)
Command Decoders (-19)
Programmer Clock (-20)
(Flight)
Signal Conditioning (-21)
PCM Encoder (-22)
Telemetry Transmitter (-23)
Data Storage Assembly (-24)
N2 Press Transducer (-25)
Turnstile Antenna (-26)
Antenna Coupler (-27)
Data Patch Unit (-29)
Dipole Antenna (-45)
Dipole Antenna (-46)
Range Rate Transponder (-47)
Power Control Assembly (-30)
Solar Cells Panel (-31)

1	2	3	4	5	6
					Battery Assembly (-32) Power Patch Unit (-33) Flight Experiment Pack Cover (-34) Flight Equipment Section (-35) Flight Aft Section (-36) Flight Support Ring (-37) Umbilical Connector Receptacles Flight Data Processor (-11)
					Subsystem Acceptance Tests
					Attitude Control Subsystem Thermal Control via Vacuum/Elec- trical Performance Thermal Control Fluid/Line Accep- tance via Leak and Vibration Command and Telemetry Power
					Thermal Control AGE Development/Qualification Tests (as required)
					AGE Functional Acceptance
					Ground System Verification
					Power Pneumatics and Fluids Command and Telemetry
					System Verification
					Balance Radio Frequency Interference/ Electromagnetic Thermal





1 2 3 4 5 6

System Engineering and Integration

Mission/Trajectory Analyses and Planning

Activation and Installation

Integrated System Test Specifications and Planning

Interface Specifications

Performance Specifications and Design

Fabrication Specifications and Engineering

Tradeoff Analyses

Reliability and Quality Assurance

Special Test Tool/Fixture Planning and Engineering

Engineering Administration, Release, and Document Control

1 2 3 4 5 6

Mission Support

- Engineering Liaison Services
- Launch Support/Checkout Operations
- Orbital Support/Experiment Operations
- Data Analyses and Processing
- Administrative Services
- Ground System(s) Checkout/Maintenance

1	2	3	4	5	6
		┆	┆		
		System/Project Management			
		Program Management			
		Mission/Operations Management			
		Design/Configuration Management			
		Data Management			

3.0 PROGRAM PLAN

The Program Plan, Figure F-1, consists of six consecutive foldout sheets interrelated by means of serially numbered connectors □ on either of the long ends. The program's hard/software services, and integration activities have been grouped into seven major strata or layers that appear throughout the six foldout sheets; namely, Stabilization/Attitude Control, Electrical, Thermal Control, Instrumentation/RF, Facilities/Mission Operations, and a systems management and integration stratum termed Spacecraft System. Activities or the elements of program work are presented as explained by the legend that appears on the first foldout sheet. Note that triangle symbols △ are Major Program Milestones and possess a number that likewise appears immediately to the right of the day-month-year calendar date on the Milestone Schedule, Table F-2. This identification permits the calendar date associated with each major program milestone to be correlated directly with the Program Plan.

The level of descriptive detail presented for each activity of the Program Plan has been established by summarizing much of the working detail that contributed to the Plan's justification and reiterative development. Although of a summary nature, these descriptions capture the sequence, essence, and philosophy of the program for Program Management deliberation and tradeoffs. The Plan is thus intended to provide a level of abstraction from which further reviews and consideration of such items as design, testing, spares, tooling, manufacture, facility usage, quality, logistics, and interfaces with other agencies may be judged during detail design progress. It should be noted with regard to the interfaces associated with the program, that the program planning was accomplished with specific consideration for the Scout launch vehicle. Although the Program Plan treats Bioresearch Module development and operations, interfaces with Scout operations are shown as submittals or acceptances of specific documentation currently in use within the Scout Program as well as participation within Mission Working Groups. The Plan thus portrays a totally integrated program consisting of launch vehicle, spacecraft, and experiment or payloads but as seen from the aspect of spacecraft integration.

The Program Plan is thus a vantage point from which to perform further management tradeoffs as necessary to arrive at a more detailed and justifiable program baseline having the flexibility for predicting/admitting authorized changes experienced in the program

course but still having the rigidity for ensuring the course to be true and manageable.

The following description is presented of the six pages of Program Plan. Sheet 1 identifies program go-ahead and the system specification and program management plan being finalized. Activities involving definition of subsystem requirements and associated analyses lead to the issuance of development specifications and the release of design engineering. These activities coupled with the preparation of a proposed document tree to present the planned arrangement of all specifications and engineering drawings, a Reliability and Quality Assurance Plan, a Safety Plan, an Activation Plan, an EMI Plan, and a baseline version of the Integrated System Test Specification lead to a Preliminary Design Review approximately three months after go ahead. This initial activity serves to coagulate the program baseline philosophies and to identify the requirements and criteria for subsystem design and development testing. Note the attention to the evolution and role of the Integrated System Test Specification in its control of the integrated system.

Sheet 2 of Figure F-1 presents the development engineering being finalized resultant from the Preliminary Design Review and the performance of efforts necessary for engineering development testing such as development tooling and the fabrication of development hardware. Also shown is the initiation of the engineering and Product Specifications that will be necessary for the development of qualification and ultimately flight hardware. Also during the period shown, the AGE specifications emerge in initial form. More specific plans are prepared for the areas of Maintenance and Repair, Logistics, Manufacturing, and the mission itself in the form of a baseline Mission and Check-out Directive which is foreseen to provide top policy guidance for preparation of the actual Directive as well as to state particular ground rules and mission guidelines. Maintainability analyses encompassing all aspects of maintenance, operability, and serviceability for both airborne and ground equipments can be seen supporting the preparation of a proposed joint use listing and the first listing of a proposed spares list and ultimately the Manufacturing Plan itself.

Sheet 3 of Figure F-1 identifies the finishing touches necessary to conduct a Critical Design Review. Specifically, tooling planning and engineering, qualification hardware engineering and Product Specifications as well as component acceptance procedures are completed. Joint Use Negotiations are completed so as to finalize what

may be shared. The input to the Scout -required Payload Description Document is compiled and submitted. A mockup becomes available for use at the Critical Design Review. Following the CDR, the actual fabrication of qualification hardware begins. Likewise, Mission Objectives and Requirements for Spacecraft No. 1 are released in preliminary form to provide the requirements for the first mission. Based on this, the Payload Mission Requirements are compiled and submitted to Scout via the Mission Working Group. An experiment simulator can be seen to support Thermal Control Qualification Tests. Following Qualification Testing, engineering and Product Specifications are shown to be updated accordingly and the fabrication of the first flight article commences. Also, the AGE for thermal control protection of the experiment is shown to undergo development and qualification testing.

Sheet 4 identifies the release and approval of the Mission Directive for Spacecraft No. 1. AGE can be seen undergoing fabrication within each of the subsystems and following its assembly, the various functional acceptance tests are performed as required. The AGE coupled with its related Product Specifications becomes available and is installed for total ground system verification and checkout at the VMSC systems test facility. Various other inputs are submitted to the Scout program in anticipation of the forthcoming first flight. For instance, mission motor data is submitted for preparation of the solid motors; an update of the Payload Description Document transmits the latest payload requirements and characteristics now that the experiments have been defined. Also, the umbilical requirements for wiring are submitted. Training of personnel and experimenter agencies is completed. Flight subsystems are assembled and acceptance tested. The first article is built up through subsystem functional tests and acceptance until finally spacecraft system level is achieved. At this time the spacecraft undergoes total system testing for approximately two months. In the meantime the launch site facilities are shown to be undergoing preparation, and AGE peculiar to launch or mission control sites is shipped.

On Sheet 5, both Spacecraft No. 1 and the AGE are shipped to Wallops Island for checkout, acceptance of a simulated experiment package for system checkout and mating with the Scout, and then launch with the flight experiment package. Note that the requirements for the second Spacecraft mission are becoming available and the Baseline Mission Objectives and Requirements for Spacecraft No. 2 are being issued. The engineering and Product Specifications are then reviewed in light of the No. 2 mission to assure design compatibility. It is well

to point out that although the baseline spacecraft will be designed to meet a basic set of mission requirements, this review loop has been used to assure complete compatibility with specific mission requirement. Any necessary revisions are made to the engineering and specifications to accommodate the second mission. The fabrication of any necessary modifications to AGE parallels fabrication of the second spacecraft. Mission No. 1 is completed approximately one month prior to the final release of the approved Mission Directive and for Spacecraft No. 2. The AGE is foreseen to be of two types: that which will be retained at Wallops Island and the mission control site or that to be recycled for checkout of the second spacecraft at the plant. Accordingly, modifications of the AGE will be either made in the plant after the AGE is recycled with modification engineering shipped to the field for installation of the change hardware.

Sheet 6 shows the AGE being reverified for Spacecraft No. 2 and the subsystem acceptances and final system testing for the second spacecraft system. It is well to note again how the Integrated System Test Specification is updated to provide the detail control for total system integration. The vertical wavy line refers to a reiterative process to accommodate the third and final spacecrafts by having the reader cycle back to essentially the middle of Sheet 5 and coming through the flow again from there for each Spacecraft until the No. 4 article at which time contract settlement would occur after the final reports and all hardware disposition had been accomplished. Milestone numbers for the third and final Spacecrafts appear beneath the Major Milestone triangles where applicable within parenthesis; the first number refers to the third Spacecraft and the second or other number refers to the fourth and final Spacecraft.

4.0 BIORESEARCH MODULE

MAJOR PROGRAM MILESTONE SCHEDULE

A printout of the calendar date by day, month, and year appears first and is immediately followed by a three character number (eg. A00) which is the milestone number. A brief description of the milestone completes the line.

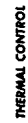
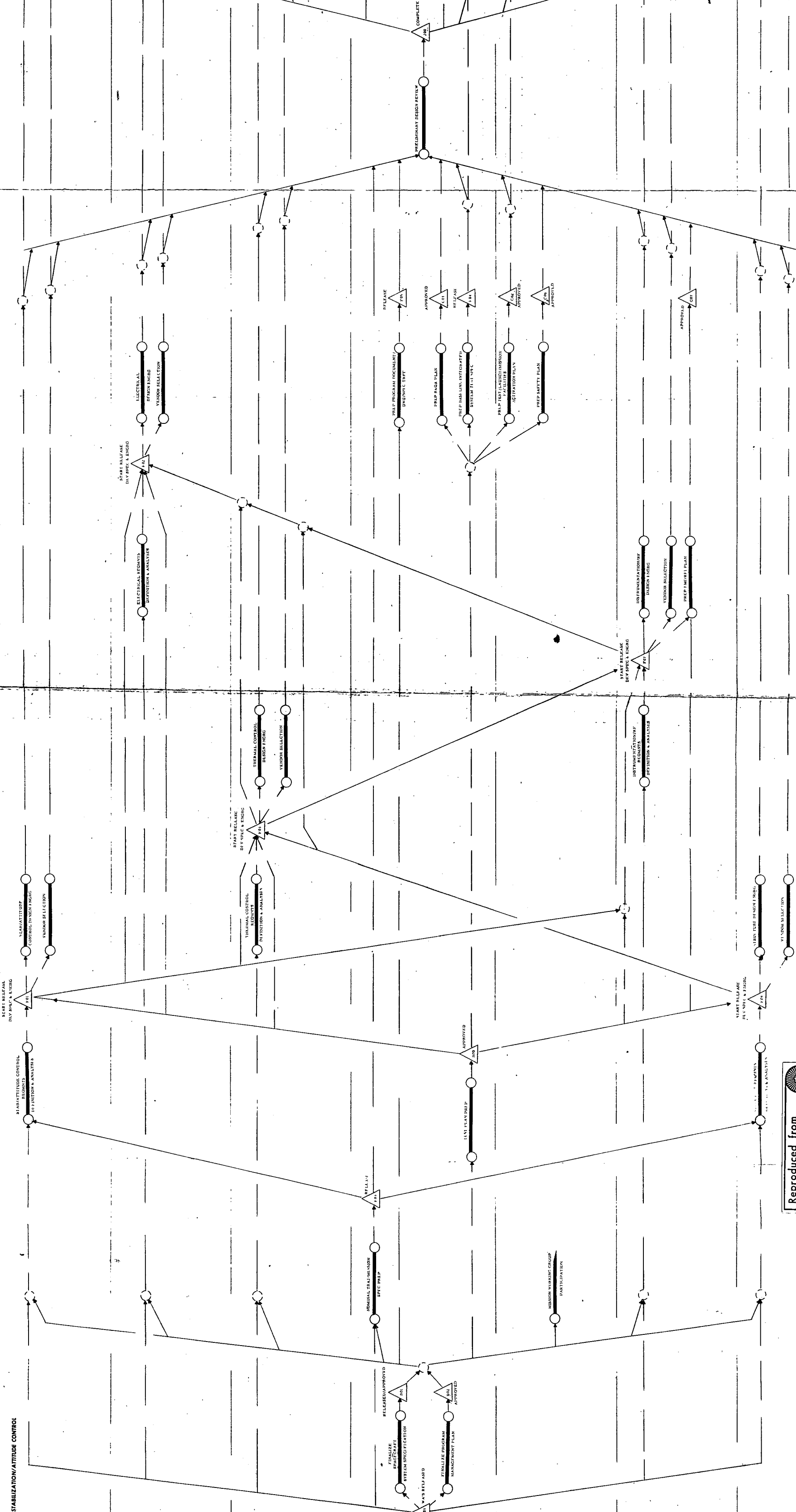
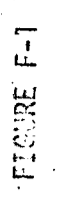
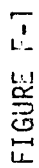


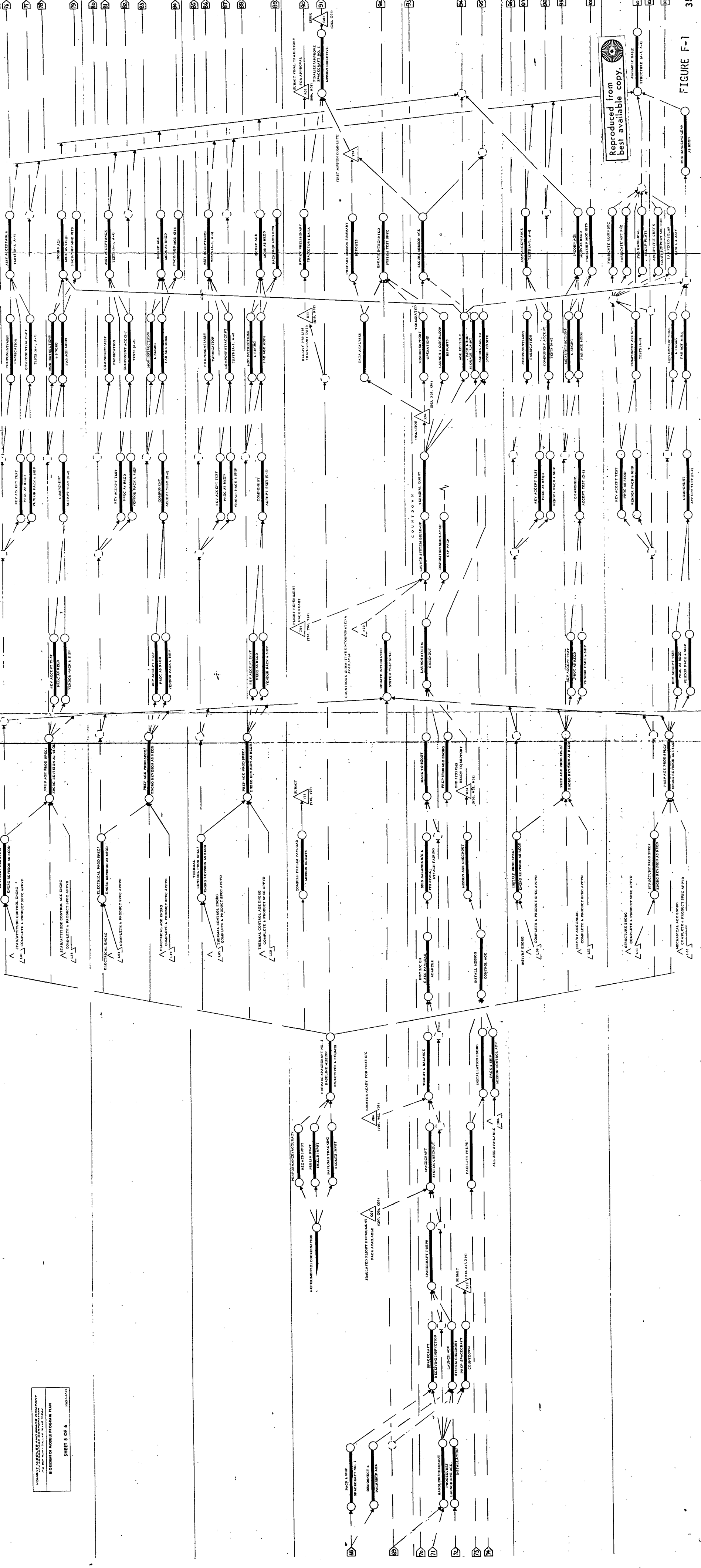
FIGURE F-7

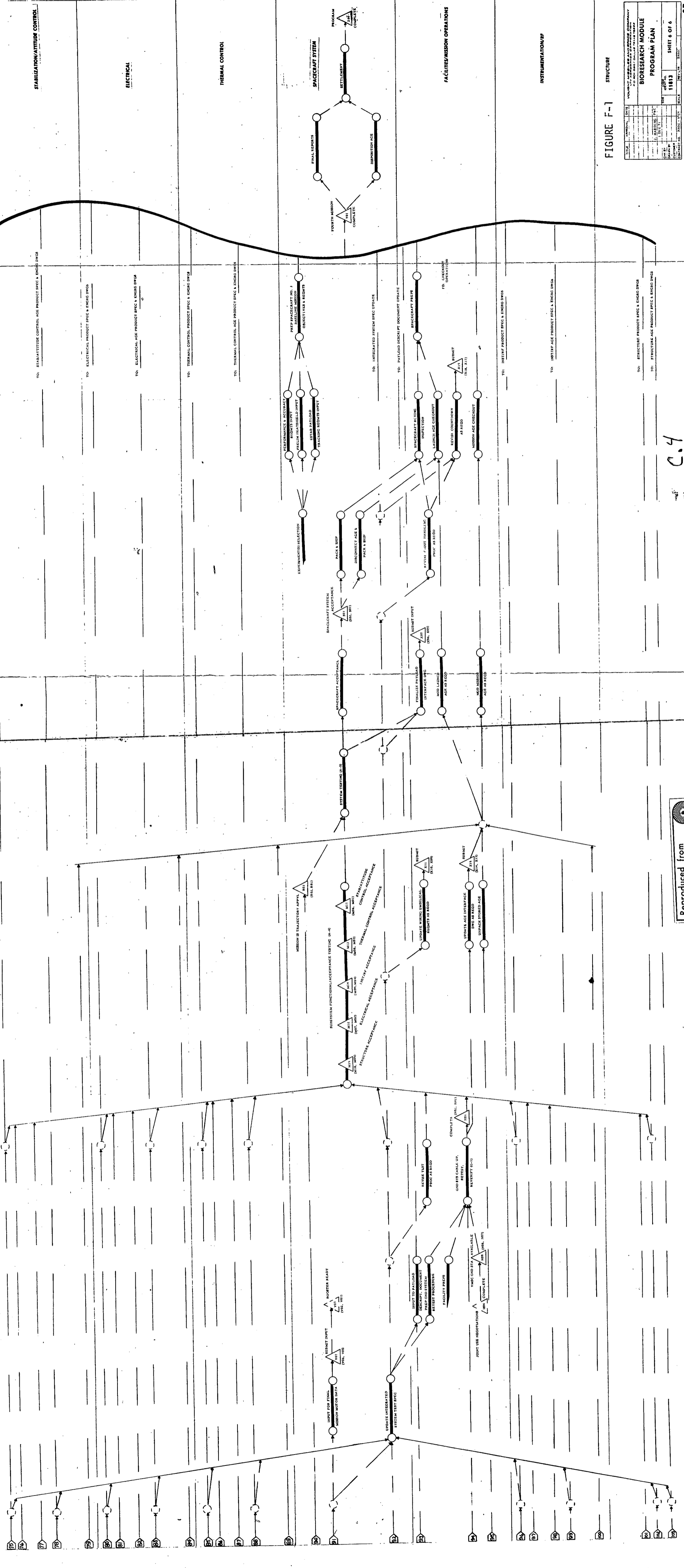






VOUGHT-VEICOR CORPORATION
AUGUST 1964
BIOHazard Model Program Plan
SHEET 5 OF 6
HALL-1414





BIORESEARCH MODULE MAJOR PROGRAM MILESTONE SCHEDULE

ISSUE - REV/ 9 DEC 71

DATE DA YR MO	MILESTONE DESCRIPTION
010672A00	PROGRAM GO AHEAD
090672B03	PROGRAM BUDGETS/SOWS/SCHEDULES RELEASED VIA WA'S
160672B02	BIORESEARCH MODULE PROGRAM MANAGEMENT PLAN APPROVED
230672B01	SPACECRAFT SYSTEM SPECIFICATION RELEASE - APPROVED
300672F09	NOMINAL TRAJECTORY - MISSION SPECIFICATION RELEASE
210772B00	BIORESEARCH MODULE TEST PLAN APPROVED
210772FC1	START RELEASE DEV ENGRG/DWGS/SPEC - STAB/ATTITUDE CONTROL
210772F08	START RELEASE DEV ENGRG/DWGS/SPEC - STRUCTURE
040872F03	START RELEASE DEV ENGRG/DWGS/SPEC - THERMAL CONTROL
090872F05	PROGRAM DOCUMENT/SPEC/DWG TREE RELEASE
110872F07	START RELEASE DEV ENGRG/DWGS/SPEC - INSTRUMENTATION/RF
230872F04	BASELINE INTEGRATED SYSTEM TEST SPEC RELEASE
250872C07	EMI/RFI PLAN APPROVED
250872C08	SAFETY PLAN APPROVED
250872F02	START RELEASE DEV ENGRG/DWGS/SPEC - ELECTRICAL
280872C02	TEST, LAUNCH, AND MISSION FACILITIES ACTIVATION PLAN APPROVED
280872C03	PROGRAM R/QA PLAN APPROVED
010972J00	PRELIMINARY DESIGN REVIEW COMPLETE
301072D01	START COLD PLATE MATERIAL/PROCESS DEVELOPMENT TESTS
101172L12	BLINE REL FOR CDR/QUAL H/W-STRUCTURE ENGRG AND PROD SPEC
101172L17	BLINE REL FOR CDR/QUAL H/W-STRUCTURE AGE/HANDLING GEAR ENGRG/PROD SPEC
171172G00	BASIC ISSUE MISSION AND CHECKOUT DIRECTIVE
171172L02	BLINE REL FOR CDR/QUAL H/W-STAB/ATTITUDE CONTROL ENGRG AND PROD SPEC
241172L06	BLINE REL FOR CDR/QUAL H/W-THERMAL CONTROL ENGRG AND PROD SPEC
241172L10	BLINE REL FOR CDR/QUAL H/W-INST/RF ENGRG AND PROD SPEC
291172L13	BLINE REL FOR CDR/QUAL H/W-STAB/ATTITUDE CONTROL AGE ENGRG/PROD SPEC
081272L04	BLINE REL FOR CDR/QUAL H/W-ELECTRICAL ENGRG AND PROD SPEC

151272L15 BLINE REL FOR CDR/QUAL H/W-THERMAL CONTROL AGE ENGRG AND PROD SPEC
 151272L16 BLINE REL FOR CDR/QUAL H/W-INST/RF AGE ENGRG AND PROD SPEC
 221272C05 MAINTENANCE AND REPAIR PLAN APPROVED
 291272L14 BLINE REL FOR CDR/QUAL H/W-ELECTRICAL AGE ENGRG AND PROD SPEC
 050173C04 LOGISTICS PLAN APPROVAL
 090273C06 MANUFACTURING PLAN APPROVED
 090373F10 SUBMIT INPUT/INTERFACE DATA TO PAYLOAD DESCRIPTION DOCUMENT
 090373J06 DEVELOPMENT TESTING COMPLETE - STRUCTURE
 160373J01 DEVELOPMENT TESTING COMPLETE - STAB/ATTITUDE CONTROL
 230373J02 DEVELOPMENT TESTING COMPLETE - ELECTRICAL
 230373J03 DEVELOPMENT TESTING COMPLETE - THERMAL CONTROL
 230373J05 DEVELOPMENT TESTING COMPLETE - INSTRUMENTATION/RF
 300373C01 PROGRAM/EXPERIMENTER TRAINING PLAN APPROVAL
 300373DC2 SPACECRAFT MOCKUP AVAILABLE FOR DESIGN REVIEW
 300373F06 TEST/LAUNCH/MISSION FACILITIES REQMTS-SPEC RELEASE
 300373F11 SPARES AND JOINT USAGE LISTING/INDEX AVAILABLE - PROPOSED
 060473M00 CRITICAL DESIGN REVIEW COMPLETE
 011073QC1 EXPERIMENT SIMULATOR AVAILABLE - FOR VMSC QUAL TESTING
 010274H04 JOINT USE NEGOTIATIONS COMPLETE
 220274K01 STAB/ATTITUDE CONTROL QUAL TEST COMPLETE
 220274K03 THERMAL CONTROL QUAL TEST COMPLETE
 220274Q04 SPARES PROVISIONING COMPLETE
 010374K09 THERMAL CONTROL AGE DEV/QUAL TEST COMPLETE
 150374K02 ELECTRICAL QUAL TEST COMPLETE
 150374K05 INSTRUMENTATION/RF QUAL TEST COMPLETE
 010474K06 STRUCTURE QUAL TEST COMPLETE
 050474V12 RELEASE PRELIMINARY PAYLOAD MISSION REQMTS - S/C NO. 1
 030574L01 STAB/ATTITUDE CONTROL PROD SPEC APPROVED AND ENGRG COMPLETE
 060574R12 RECEIVE PRELIMINARY TRAJECTORY DATA - S/C NO. 1
 100574L05 THERMAL CONTROL PROD SPEC APPROVED AND ENGRG COMPLETE
 170574L20 THERMAL CONTROL AGE PROD SPEC APPROVED AND ENGRG COMPLETE
 240574L09 INSTRUMENTATION/RF PROD SPEC APPROVED AND ENGRG COMPLETE
 310574L18 STAB/ATTITUDE CONTROL AGE PROD SPEC APPROVED AND ENGRG COMPLETE
 140674L03 ELECTRICAL PROD SPEC APPROVED AND ENGRG COMPLETE
 210674L21 INSTRUMENTATION/RF AGE PROD SPEC APPROVED AND ENGRG COMPLETE
 280674L19 ELECTRICAL AGE PROD SPEC APPROVED AND ENGRG COMPLETE
 280674R08 SUBMIT FINAL TRAJECTORY ROMAN 1 * FOR APPROVAL - S/C NO.1

110774L11 STRUCTURE SUBSYSTEM PROD SPEC APPROVED AND ENGRG COMPLETE
 180774L22 STRUCTURE AGE /HANDLING GEAR PROD SPEC APPROVED AND ENGRG COMPLETE
 290774N05 S/C SYSTEM TEST FACILITY AVAILABLE FOR AGE INSTALLATION
 290774N06 ALL AGE AVAILABLE
 050874X12 INPUT WIRING UMBILICAL REQUIREMENTS - S/C NO. 1
 090874G04 FINALIZE AND APPROVE SPACECRAFT NO. 1 MISSION DIRECTIVE
 120874N10 VMSC GROUND STATION AVAILABLE FOR CHECKOUT OF S/C NO. 1
 230874Q02 TRAINING COURSES COMPLETE
 230874Q03 FLIGHT SPARES AIRBORNE AND AGE AVAILABLE* AVAILABLE
 020974N04 GND SYS/FAC CHECKOUT COMPLETE AT VMSC FOR SYS TEST S/C NO. 1
 020974Q00 START BUILD-UP OF S/C NO.1 TO SYSTEM LEVEL
 020974V08 FINAL MISSION MOTOR DATA SUBMITTED - S/C NO. 1
 120974M20 STRUCTURE SUBSYSTEM ACCEPTANCE - S/C NO. 1
 190974M17 ELECTRICAL SUBSYSTEM ACCEPTANCE - S/C NO. 1
 260974M19 INSTRUMENTATION /RF SUBSYSTEM ACCEPTANCE - S/C NO.1
 071074M18 THERMAL CONTROL SUBSYSTEM ACCEPTANCE - S/C NO. 1
 101074M16 STAB/ ATTITUDE CONTROL SUBSYSTEM ACCEPTANCE - S/C NO.1
 111074R04 MISSION ROMAN 1 TRAJECTORY APPROVAL - S/C NO. 1
 041174X16 INPUT TO AGE INTERFACE DRAWING SUBMITTED - S/C NO.1
 271174X08 INPUT TO PAYLOAD INTERFACE DRAWING SUBMITTED - S/C NO. 1
 061274S04 S/C NO.1 SYSTEM ACCEPTANCE - FACI COMPLETE
 161274W04 LAUNCH/MISSION GROUND SYSTEMS READY TO SUPPORT S/C NO.1
 301274Q08 SIMULATED EXPERIMENT PACK AVAIL FOR CHECKOUT OF S/C NO.1 - FIELD
 030175T04 FLIGHT EXPERIMENT PACK READY FOR INSERTION IN S/C NO. 1
 150175X20 SUBMIT S/C COUNTDOWN INPUT TO COUNTDOWN DOCUMENT - S/C NO. 1.
 170175Y04 BOOSTER READY FOR S/C NO.1
 010275X04 S/C NO.1 INSERTION - START MISSION OPERATIONS
 040475V11 RELEASE PRELIMINARY PAYLOAD MISSION REQMNIS - S/C NO. 2
 050575R11 RECEIVE PRELIMINARY TRAJECTORY DATA - S/C NO.2
 010675Y04 FIRST MISSION SUPPORT COMPLETE
 270675R07 SUBMIT FINAL TRAJECTORY ROMAN ONE S* FOR APPROVAL - S/C NO.2
 040875X11 INPUT WIRING UMBILICAL REQUIREMENTS - S/C NO. 2
 080875G03 FINALIZE AND APPROVE SPACECRAFT NO. 2 MISSION DIRECTIVE
 110875N09 VMSC GROUND STATION AVAILABLE FOR CHECKOUT OF S/C NO. 2
 010975N03 GND SYS/FAC CHECKOUT COMPLETE AT VMSC FOR SYS TEST S/C NO. 2
 010975V07 FINAL MISSION MOTOR DATA SUBMITTED S/C NO. 2
 190975M15 STRUCTURE SUBSYSTEM ACCEPTANCE - S/C NO. 2
 250975M12 ELECTRICAL SUBSYSTEM ACCEPTANCE - S/C NO. 2

091075M14 INSTRUMENTATION/RF SUBSYSTEM ACCEPTANCE - S/C NO. 2
 101075R03 MISSION ROMAN ONE S TRAJECTORY APPROVAL - S/C NO. 2
 131075M13 THERMAL CONTROL SUBSYSTEM ACCEPTANCE - S/C NO. 2
 241075M11 STAB/ATTITUDE CONTROL SUBSYSTEM ACCEPTANCE - S/C NO. 2
 031175X15 INPUT TO AGE INTERFACE DRAWING SUBMITTED - S/C NO. 2
 261175X07 INPUT TO PAYLOAD INTERFACE DRAWING SUBMITTED - S/C NO. 2
 131275S03 SPACECRAFT SYSTEM ACCEPTANCE - S/C NO. 2
 291275Q07 SIMULATED EXPERIMENT PACK AVAIL FOR CHECKOUT OF S/C NO. 2 - FIELD
 291275W03 LAUNCH/MISSION GROUND SYSTEMS READY TO SUPPORT S/C NO. 2
 050176T03 FLIGHT EXPERIMENT PACK READY FOR INSERTION IN S/C NO. 2
 090176V03 BOOSTER READY FOR S/C NO. 2
 150176X10 SUBMIT S/C COUNTDOWN INPUT TO COUNTDOWN DOCUMENT - S/C NO. 2
 010276X03 S/C NO. 2 INSERTION - START MISSION OPERATIONS
 020476V10 RELEASE PRELIMINARY PAYLOAD MISSION REQMENTS - S/C NO. 3
 030576R10 RECEIVE PRELIMINARY TRAJECTORY DATA - S/C NO. 3
 010676Y03 SECOND MISSION SUPPORT COMPLETE
 250676R06 SUBMIT FINAL TRAJECTORY ROMAN TWO* FOR APPROVAL - S/C NO. 4
 020876X10 INPUT WIRING UMBILICAL REQUIREMENTS - S/C NO. 3
 060876G02 FINALIZE AND APPROVE SPACECRAFT NO. 3 MISSION DIRECTIVE
 130876N08 VMSC GROUND STATION AVAILABLE FOR CHECKOUT OF S/C NO. 3
 060976N02 GND SYS/FAC CHECKOUT COMPLETE AT VMSC FOR SYS TEST S/C NO. 3
 060976V06 FINAL MISSION MOTOR DATA SUBMITTED S/C NO. 3
 170976M10 STRUCTURE SUBSYSTEM ACCEPTANCE - S/C NO. 3
 240976M07 ELECTRICAL SUBSYSTEM ACCEPTANCE - S/C NO. 3
 081076R02 MISSION ROMAN TWO TRAJECTORY APPROVAL - S/C NO. 3
 111076M09 INSTRUMENTATION/RF SUBSYSTEM ACCEPTANCE - S/C NO. 3
 181076M08 THERMAL CONTROL SUBSYSTEM ACCEPTANCE - S/C NO. 3
 011176X14 INPUT TO AGE INTERFACE DRAWING SUBMITTED - S/C NO. 3
 051176M06 STAB/ATTITUDE CONTROL SUBSYSTEM ACCEPTANCE - S/C NO. 3
 241176X06 INPUT TO PAYLOAD INTERFACE DRAWING SUBMITTED - S/C NO. 3
 191276S02 SPACECRAFT SYSTEM ACCEPTANCE - S/C NO. 3
 271276Q06 SIMULATED EXPERIMENT PACK AVAIL FOR CHECKOUT OF S/C NO. 3 - FIELD
 271276W02 LAUNCH/MISSION GROUND SYSTEMS READY TO SUPPORT S/C NO. 3
 040177T02 FLIGHT EXPERIMENT PACK READY FOR INSERTION IN S/C NO. 3
 140177V02 BOOSTER READY FOR S/C NO. 3
 150177X18 SUBMIT S/C COUNTDOWN INPUT TO COUNTDOWN DOCUMENT - S/C NO. 3
 010277X02 S/C NO. 3 INSERTION - START MISSION OPERATIONS

010477V09 RELEASE PRELIMINARY PAYLOAD MISSION REQMENTS - S/C NO. 4
 020577R09 RECEIVE PRELIMINARY TRAJECTORY DATA - S/C NO. 4
 010677Y02 THIRD MISSION SUPPORT COMPLETE
 240677R05 SUBMIT FINAL TRAJECTORY ROMAN TWO* FOR APPROVAL - S/C NO. 4
 010877X09 INPUT WIRING UMBILICAL REQUIREMENTS - S/C NO. 4
 050877G01 FINALIZE AND APPROVE SPACECRAFT NO. 4 MISSION DIRECTIVE
 190877N07 VMSC GROUND STATION AVAILABLE FOR CHECKOUT OF S/C NO. 4
 050977N01 GND SYS/FAC CHECKOUT COMPLETE AT VMSC FOR SYS TEST S/C NO. 4
 050977V05 FINAL MISSION MOTOR DATA SUBMITTED S/C NO. 4
 160977M05 STRUCTURE SUBSYSTEM ACCEPTANCE - S/C NO. 4
 230977M02 ELECTRICAL SUBSYSTEM ACCEPTANCE - S/C NO. 4
 071077R01 MISSION ROMAN TWO TRAJECTORY APPROVAL - S/C NO. 4
 171077M04 INSTRUMENTATION/RF SUBSYSTEM ACCEPTANCE - S/C NO. 4
 241077M03 THERMAL CONTROL SUBSYSTEM ACCEPTANCE - S/C NO. 4
 031177X13 INPUT TO AGE INTERFACE DRAWING SUBMITTED - S/C NO. 4
 041177M01 STAB/ATTITUDE CONTROL SUBSYSTEM ACCEPTANCE - S/C NO. 4
 221177X05 INPUT TO PAYLOAD INTERFACE DRAWING SUBMITTED - S/C NO. 4
 171277S01 SPACECRAFT SYSTEM ACCEPTANCE - S/C NO. 4
 261277Q05 SIMULATED EXPERIMENT PACK AVAIL FOR CHECKOUT OF S/C NO. 4 - FIELD
 271277W01 LAUNCH/MISSION GROUND SYSTEMS READY TO SUPPORT S/C NO. 4
 050178T01 FLIGHT EXPERIMENT PACK READY FOR INSERTION IN S/C NO. 4
 130178V01 BOOSTER READY FOR S/C NO. 4
 150178X17 SUBMIT S/C COUNTDOWN INPUT TO COUNTDOWN DOCUMENT - S/C NO. 4
 010278X01 S/C NO. 4 INSERTION - START MISSION OPERATIONS
 010678Y01 FOURTH MISSION SUPPORT COMPLETE
 290978Z00 PROGRAM COMPLETE

SUMMARY

(To be used in preparation of Library-Card Abstract)

Preliminary designs of the Bioexplorer spacecraft, developed in an earlier study program, are analyzed and updated to conform to a new specification which includes use of both the Scout and the Space Shuttle Vehicle for launch. The new spacecraft design, referred to as Bioresearch Module, is capable of supporting a variety of small biological experiments in near-earth and highly elliptical earth orbits. Inboard profile drawings, weight statements, interface drawings, and equipment lists are provided to document the design. Considerable study is devoted to use of the Space Shuttle Vehicle for launch and retrieval. It is shown that the Bioresearch Module spacecraft is compatible with both Scout and the Space Shuttle Vehicle, but that the latter requires additional on-board hardware for launch support. A development plan is included to provide data for planning subsequent phases of the Bioresearch Module program.